

For presentation in South Africa
on Oct. 29th, 1963

FACILITY FORM 802

N65-89113
(ACCESSION NUMBER)
40
(PAGES)
CR-74840
(NASA CR OR TMX OR AD NUMBER)

(THRU)
(CODE)
(CATEGORY)

SOME APPLICATIONS OF ENGINEERING MECHANICS TO THE DEVELOPMENT OF SPACE VEHICLES¹

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INTRODUCTION

The program of space exploration upon which the United States has recently embarked provides many challenges and opportunities in nearly every field of engineering. Giant boost rockets require enormous launch pads; long-range communications demand very large, highly precise, and delicately controlled antennas; precise interplanetary trajectories depend upon gyroscopes and sensors of extreme precision and sensitivity; complex electronic control systems must not only be of minimum size and weight, but must also function with unparalleled reliability; and spacecraft structures must provide adequate strength with absolute minimum weight. Thus, in every field, engineers must develop new technology and find improved solutions to design problems.

At the same time, because of the emphasis necessarily placed upon system integration and optimization, each engineering discipline must be applied in the environment of technical problems associated with the other disciplines. We find electronic engineers, civil engineers, mechanical engineers, and biologists working side by side attempting to find the optimum trade-off solutions to their common problems. I believe this is healthy. Each of us is acquiring a broader outlook and learning to use new tools as a consequence of this intimate working relationship.

The actual space vehicle, or spacecraft, is only one element of the total system required

to carry out a flight mission. In some respects it is the most glamorous, since in effect it constitutes the "payload"; however, it is no more important than other elements of the system, and it may be a good deal less expensive than certain other parts, such as the launching rocket.

Because the spacecraft actually carries out the space flight itself, and because the development of spacecraft is a subject with which I have some familiarity, this paper discusses space vehicle design and development. In particular, it is intended to cover the application of engineering mechanics disciplines to three related aspects of spacecraft development: configuration, structural development, and temperature control. Emphasis is placed upon identification of constraints and requirements, general approaches to design, and upon analytical and experimental tools available to the engineer.

TYPICAL SPACE MISSION

Primarily for the purpose of establishing a perspective from which to discuss several aspects of space vehicle development, it is worthwhile to describe a typical unmanned flight mission. Missions can be expected to vary considerably in detail, depending upon specific scientific and technological objectives. However, if we confine our attention to vehicle systems employing chemical propellants, we find that most missions consist essentially of the following phases: launch and injection, initial cruise, mid-course correction, cruise, and target encounter. During the first phase, the vehicle system consists of the space vehicle, plus one or more of the launching rocket stages. During all subsequent phases the vehicle system consists only of the spacecraft.

¹This paper presents the results of one phase of research carried out at the Jet Propulsion Laboratory, California Institute of Technology, under Contract No. NAS7-100, sponsored by the National Aeronautics and Space Administration.

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The typical flight mission to be described is that of the Mariner 2 spacecraft, which had the mission objective of performing a close flyby of the planet Venus and making radiation-intensity measurements in both the microwave and infrared regions on that planet. The scientific objective was to determine the approximate surface and cloud-top temperatures prevailing on Venus. The launch vehicle consisted of a modified Atlas ballistic missile and an Agena upper stage. It was the function of the launch vehicle system to boost the spacecraft to initial conditions of velocity, both in magnitude and direction, corresponding to a ballistic trajectory to encounter with the target planet. Establishment of such initial conditions is referred to as injection. For the launch dates considered practicable, this vehicle system had the capability of so injecting about 450 pounds of payload. The Mariner 2 spacecraft was, therefore, designed to a weight limit of 450 pounds. Additional constraints upon the spacecraft, as discussed later, were established in consideration of the Agena structure and geometry, and the launch vehicle guidance accuracy. On the launch pad, the total vehicle system stood approximately 100 feet high, as illustrated in Fig. 1. At the bottom was the Atlas rocket. On top of the Atlas was the Agena, and on top of the Agena, enclosed within and protected by a jettisonable "nose cone" or "shroud", was the Mariner 2 spacecraft.

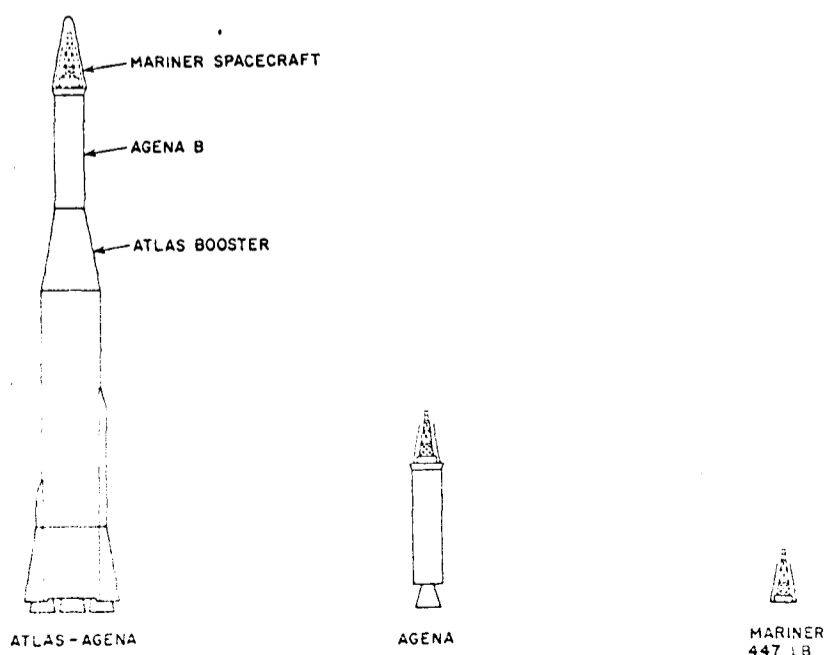


Fig. 1. Mariner 2 launch vehicle

In the launch and injection sequence, the Atlas engines are started a few seconds prior to liftoff, and the vehicle is not released for flight until full thrust is achieved on all engines. Initial flight is vertical, and the vehicle is rolled about its long axis in order that a subsequent pitch-over takes place along the proper flight path in inertial space. The Atlas is sometimes called a one-and-a-half-stage rocket. It has three rocket engines, two of which are referred to as "booster engines" and which are jettisoned after a short time. The third engine is called the sustainer engine, and burns throughout the period from liftoff to stage separation. In the Mariner 2 mission, shortly after sustainer engine cutoff, the Agena shroud, which had protected the spacecraft during the high-velocity flight out through the Earth's atmosphere, was ejected forward and away from the vehicle. The Agena was then separated from the Atlas rocket, oriented to the proper attitude relative to the Earth's horizon, and its rocket motor fired for a sufficient length of time to accelerate it to orbital velocity.

Upon reaching orbital velocity the Agena, with the Mariner on its nose, continued to coast at approximately a 100-nautical-mile altitude above the Earth's surface for approximately 20 minutes in what is termed a "parking orbit". The actual duration of flight in the parking orbit depended upon the actual time of launch. When the Agena reached the proper point in inertial space, the rocket motor was again ignited and the vehicle was accelerated to the proper velocity for the space flight initial conditions.

Shortly after Agena engine shutdown the spacecraft was separated from the Agena stage, such separation being accomplished by means of small springs under preload at the attachment points. To prevent the Agena from following the spacecraft on its flight to Venus, it was then necessary to perform an escape maneuver with the Agena. This was done by yawing the Agena rocket to a skew angle and expelling unused fuel under pressure. This gave enough impulse to cause the Agena to go off on a divergent course.

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The Mariner 2 was an attitude-controlled spacecraft, requiring Sun orientation for electrical power and antenna orientation for communication. Accordingly, after deployment of the solar panels and the antenna, the spacecraft autopilot system went into operation, stabilized the spacecraft from its residual tumbling motion, sensed the location of the Sun, and oriented the spacecraft so that the solar panels were normal to the Sun line. In this orientation the spacecraft then cruised for approximately 8 days during which precise tracking information was obtained from the antenna stations located in California, Australia, and Johannesburg.

After several days of tracking it had been determined that the miss distance at Venus for the injection conditions achieved would be approximately 250,000 miles. The velocity correction, in magnitude and direction, necessary to establish a trajectory for the desired miss distance of approximately 20,000 miles had also been calculated. Based upon this information the spacecraft was then instructed to carry out a mid-course maneuver to establish the new trajectory.

Performance of the midcourse maneuver required that the spacecraft give up its Sun and Earth orientation under autopilot control, orient its long axis to a specified inertial direction, and burn its small correction rocket motor for a specified length of time. The actual maneuver performed in the case of Mariner 2 required a roll turn of approximately 9 degrees, a 140-degree pitch turn, and motor-burning time of 27.8 seconds. Upon completion of this maneuver the spacecraft reacquired its Sun orientation and rolled to a position in which the high-gain dish antenna was aimed at the Earth. Cruise conditions thus having been restored, the spacecraft then continued to cruise for approximately 100 more days.

The Mariner 2 mission did not require reorientation of the spacecraft during the planet-encounter phase. Orientation with respect to the Sun and Earth was continuously maintained. However, instruments designed to scan the surface of the planet during the flyby were turned on and

calibrated, and changes were made in the telemetry mode to obtain a maximum amount of information from these instruments during the approximately 40-minute period of encounter. The spacecraft approached Venus from above and outside, as illustrated in Fig. 2.

Upon completion of the encounter phase the spacecraft continued in a cruising condition, maintaining communication with the Earth, for approximately another 20 days, at which time, for unknown reasons, contact was lost. The spacecraft is in a perpetual orbit about the Sun, but it is now almost certainly slowly tumbling through space, its supply of attitude-control gas exhausted, and its instruments and equipment dead. If there is a heaven for spacecraft, the Mariner 2 certainly deserves to be there, for it performed in an exemplary fashion. All systems continued to function until mission objectives had been fully achieved, important scientific information concerning Venus temperatures was returned to the Earth, and engineering information concerning the environment of space and its effects upon the spacecraft was obtained. The Mariner 2 has gone to a well-deserved rest.

With this very brief introductory description of what takes place on a typical unmanned interplanetary space flight, we can now proceed to a discussion of certain aspects of the vehicle design and development.

CONFIGURATION DESIGN

Constraints

The basic problem in configuration design can be defined as follows: Given a number of discrete system elements, each with a set of requirements, arrange these elements into a coherent design that satisfies all imposed constraints.

The problem is basically a geometrical one, but questions of structural efficiency and of manufacturing feasibility are always involved. For this reason, the configuration design must evolve along with the structural, temperature control, and electronic subsystem designs. The

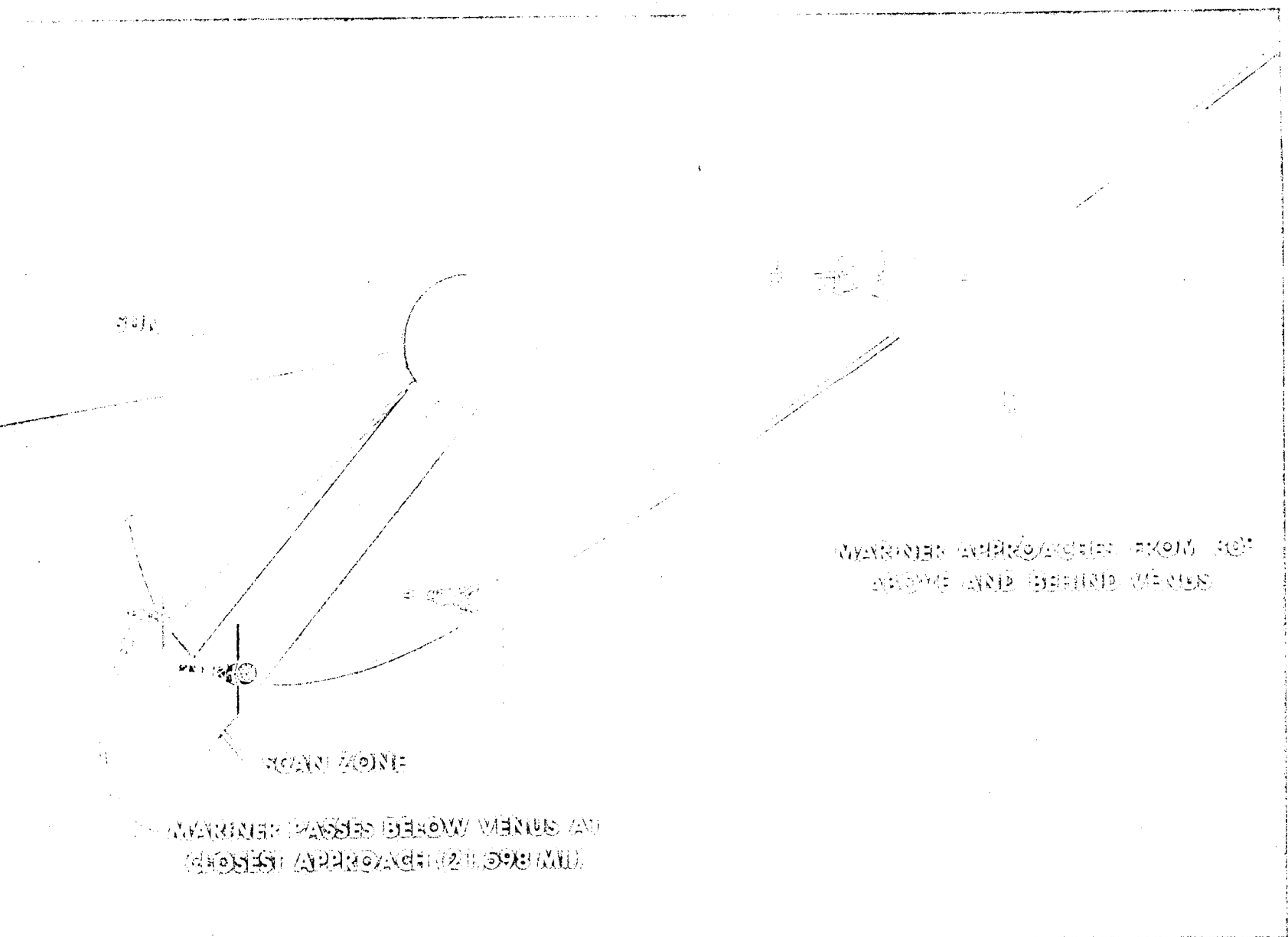


Fig. 2. Mariner 2 Venus flyby as seen from direction of Earth

problem is usually attacked in an iterative fashion, starting with an arrangement similar to what has been used for a previous vehicle. In fact, for reasons of both time and resource limitations, the vehicle design may be constrained to utilize certain existing hardware from a previous development. The elements are arranged according to some concept, and the arrangement is then tested to determine whether or not the requirements are met and the constraints satisfied. In this section of the paper it is my intention to describe some of the constraints which typically apply, and to indicate some techniques which have been developed for analyzing potential arrangements.

One of the basic constraints established, once a boost vehicle has been selected for a given mission, is the volume available inside the booster nose cone or shroud, as illustrated in

Fig. 3. The shroud is generally thought of as an integral part of the boost rocket. A special shroud might be developed for a particular mission, but a substantial development program, involving interactions with the boost rocket, would be required. If such a shroud modification is

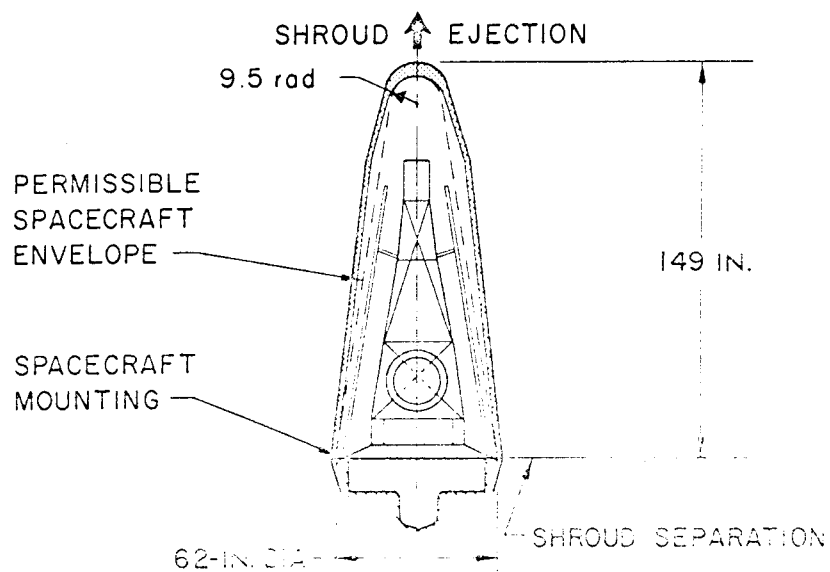


Fig. 3. Shroud geometry constraints

desired, it is relatively easy to change its length, but considerably more serious to increase its diameter, which will require "hammer heading" if the shroud diameter is to exceed that of the basic rocket stage. Since even in its folded configuration the spacecraft is expected to be a flexible structure, adequate clearance must be provided for dynamic deflections encountered during boost. In addition, allowance must be made for the motion of the shroud as it is ejected from the stage. Some shroud designs are ejected forward, while others open up like clam shells. Other constraints imposed by the boost vehicle involve the rocket upper structure and the corresponding means by which the loads must be carried into that structure.

Once the spacecraft is separated from the boost rocket there are certain geometrical constraints which relate to the system concept upon which the vehicle design is based. Several of these constraints will be briefly described.

In the case of an attitude-stabilized spacecraft, such as Mariner 2, Sun orientation is maintained, and solar panels must be arranged so as to receive solar energy at all times. The Sun then provides one required attitude reference, and Sun sensors must be so located that the Sun can be seen by one sensor regardless of the tumbling mode which may prevail. Although it is sometimes possible to use the Earth as a second attitude reference, as in the case of Mariner 2, the more general case requires that a selected star be used for this purpose. The preferred star is frequently Canopus, a very bright star in the southern hemisphere approximately 7 degrees from a normal to the ecliptic plane. We are fortunate, in fact, that a suitable star is so located, but as illustrated in Fig. 4, a sensor geometry problem still exists. The case illustrated is for a Mars mission, and it is to be noted that in the required 7 months of cruising flight with the spacecraft roll axis pointing at the Sun, the Canopus sensor angle must change by nearly 15 degrees because of angular rotation of the spacecraft in inertial space. Obviously, if the Canopus sighting angle was more than 7 degrees

from the ecliptic normal this problem would be much more severe.

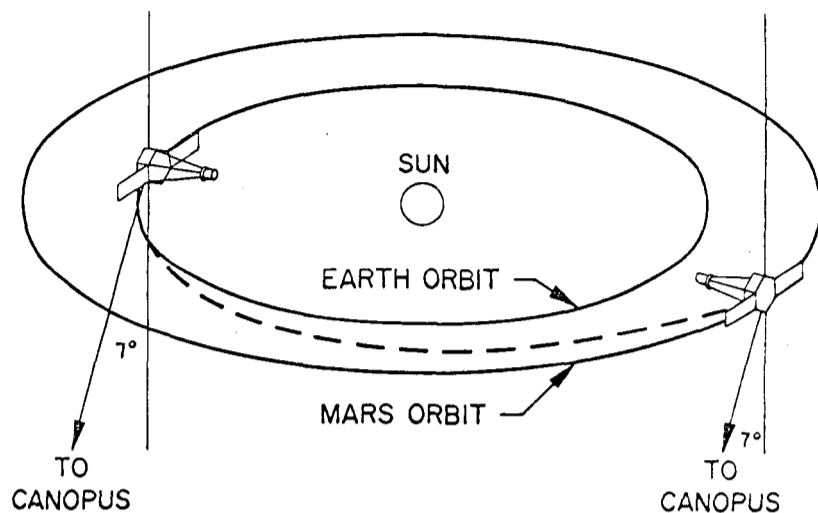


Fig. 4. Typical geometry for Mars trajectory

Almost every spacecraft communication system requires some form of directional antenna, since transmitter power is strictly limited by size, weight, and power considerations. This means that such an antenna must, in general, be articulated relative to a Sun-oriented spacecraft in order to keep the Earth in the high-gain beam. This normally introduces a fairly strong geometrical constraint on the spacecraft design.

A midcourse velocity correction typically involves the use of a small rocket motor with a thrust of approximately one-tenth the weight of the spacecraft. Even for this relatively low thrust level, however, it is essential that the thrust axis pass very close to the center of gravity in order to avoid serious problems of attitude control during burning.

From the configuration standpoint, the target-encounter of a space flight is usually both critical and difficult. Instruments must be so located that they can maintain a view of the target object, the communication antenna must have an unobstructed view of the Earth, and various sensors must not be allowed to become confused by other astronomical bodies. To understand and analyze the geometrical situation near the target planet, it is necessary to adopt the viewpoint of a hypothetical observer located on the spacecraft.

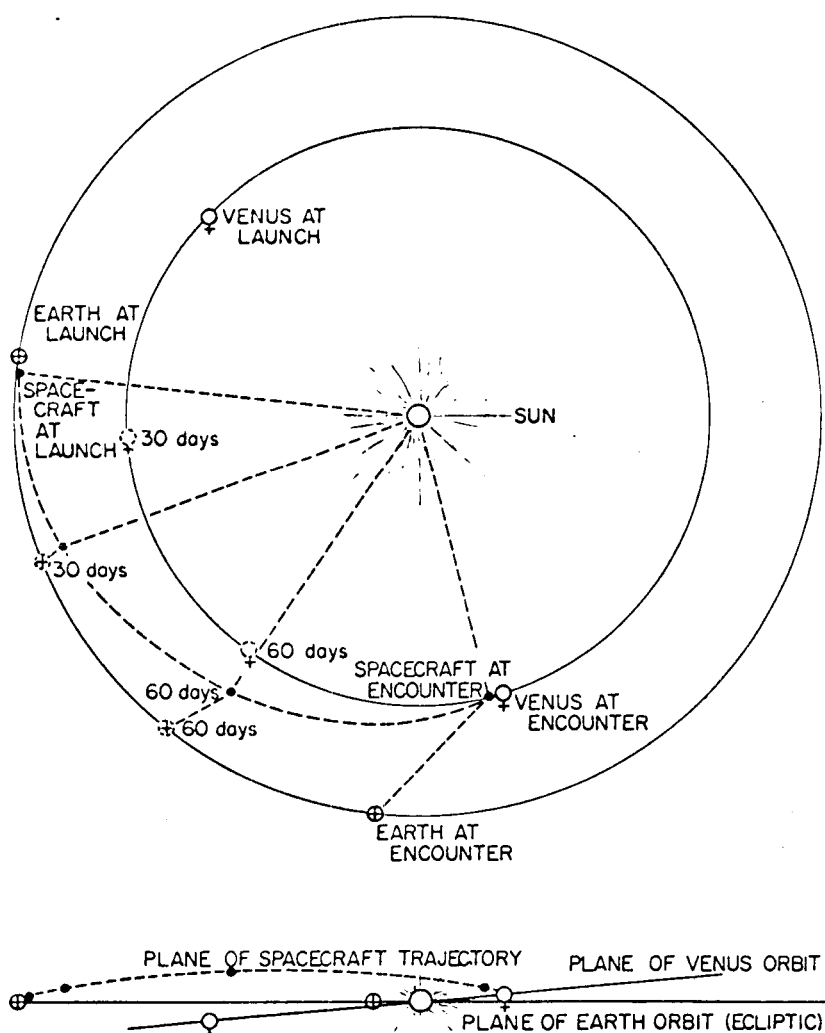


Fig. 5. Earth-Venus trajectory for Mariner 2

The trajectory for the Mariner 2 mission to Venus is shown in Fig. 5. The spacecraft was launched in such a direction as to initially fall behind the Earth in its orbit about the Sun. As the spacecraft then fell in toward the Sun it picked up speed, passed the Earth, and eventually overtook Venus from the rear and outside. The orbital planes of the Earth and Venus do not coincide, and the spacecraft followed a trajectory lying in a third plane, as illustrated in Fig. 5. Thus, in an edge view of the ecliptic plane, the spacecraft appears to rise above the ecliptic plane and then come down through the orbital plane of Venus. (The sketch is somewhat exaggerated to illustrate the point.) The angles involved are very small when velocities are measured relative to coordinates fixed in the solar system. However, the angles are by no means small when they are considered in terms of an observer fixed on the spacecraft or on the target planet, as illustrated in Fig. 6. Here we

see that the Mariner 2 appeared, from the viewpoint of an observer on Venus, to approach steeply from outside and above the planet. The relative flight path illustrated is not exaggerated.

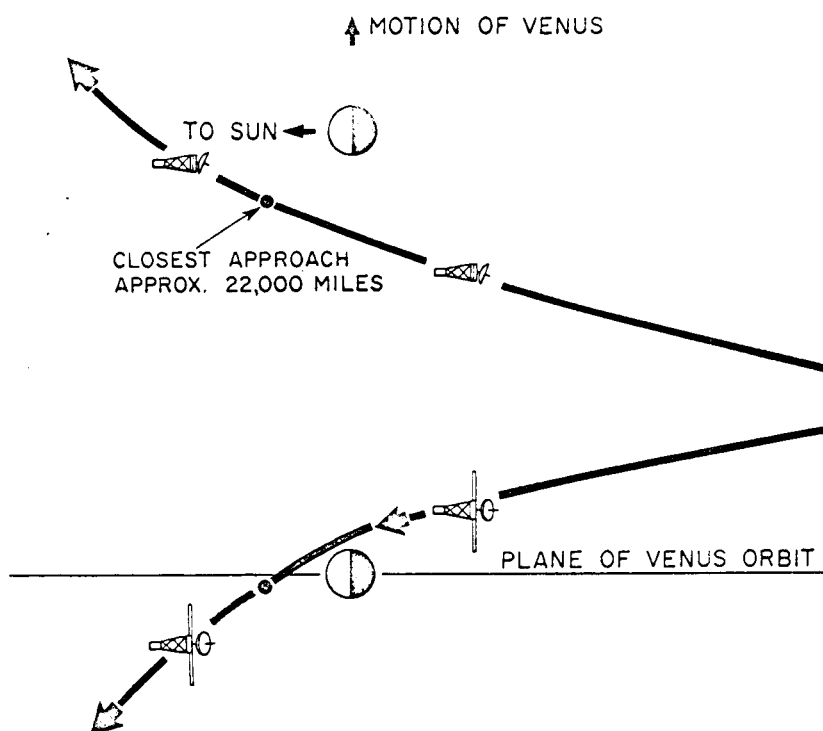


Fig. 6. Mariner 2 trajectory in Venus-fixed coordinates

It can be seen that the configuration constraints imposed near the target planet will be strongly affected by the specific near-planet trajectory or aiming point selected. Obviously, however, the mission objectives must largely control aiming-point selection, and the usual practice is to determine an acceptable aiming region which will satisfy the scientific objectives of the mission and still enable a practicable design to be established.

Figure 7 indicates some geometrical relationships and illustrates some constraints associated with the near-planet trajectory. The "incoming asymptote" simply corresponds to the flight path unperturbed by the local gravitational attraction of the target planet. Interplanetary trajectories are all established in terms of the direction of the incoming asymptote. The aiming-point plane is normal to the direction of the incoming asymptote, and the aiming point is the intersection of the incoming asymptote with

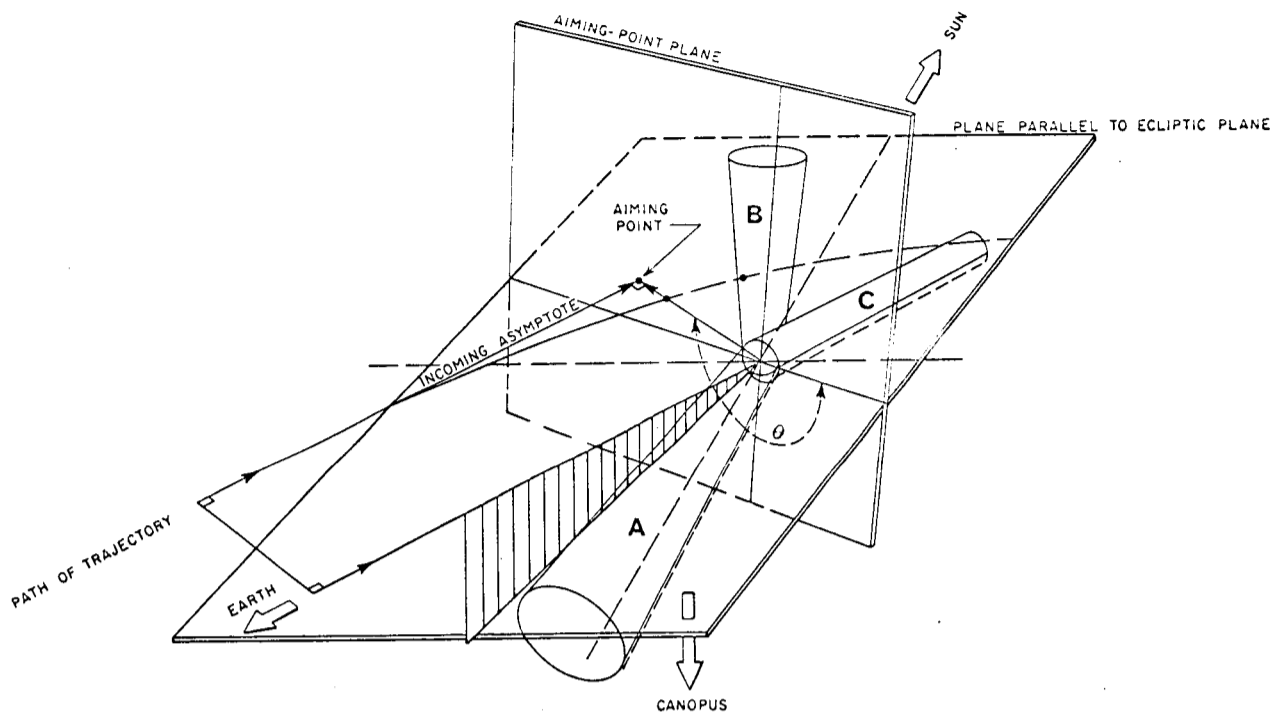


Fig. 7. Trajectory geometry and constraints near target planet

the aiming-point plane. Several forbidden zones are illustrated in Fig. 7. These correspond respectively to constraints that neither the sensors nor instruments shall confuse the planet and the Sun (A), that Canopus shall not be eclipsed by the planet (B), and that the spacecraft shall be able to see the Earth at all times (C). Since the direction of the incoming asymptote is not affected by changing the aiming point, we can now examine acceptable near-planet trajectories in terms of a two-dimensional representation in polar coordinates of the aiming-point plane. Figure 8 shows a typical aiming-point diagram for a planetary flyby mission. Aiming points in the shaded zones are not permissible. The impact area at the center is larger than the target planet and corresponds to the aiming points within which gravitational attraction will perturb the orbit sufficiently to cause impact.

Configuration Analysis

As has been mentioned previously, the development of a configuration for a particular spacecraft is significantly affected by past history and experience. Very frequently the design is approached by taking a configuration used for a prior and different mission, and modifying that configuration to conform to the specific

constraints established by the new mission. Alternatively, the new configuration may have been conceived, in general, during a previous project development at a time when major configuration changes were impossible. The new project may represent an opportunity to try out previously conceived ideas. In any case, the process is one in which the several specific elements are put together according to an educated and experienced guess, the configuration is checked to determine if all requirements are met and constraints satisfied, and adjustments are made to determine a better-educated guess.

An important facet of geometrical configuration has to do with the "look-angle" problem. This problem can briefly be described as that of permitting "looking" elements to "see" in their desired directions without interference. In its most elementary form, it involves:

1. Interference of spacecraft elements with the field of view of sensors or instruments
2. Physical interferences associated with the limited articulation of looking members

In its more sophisticated aspects, it also includes:

1. Compatibility of required tracking rates with servo or actuator capability

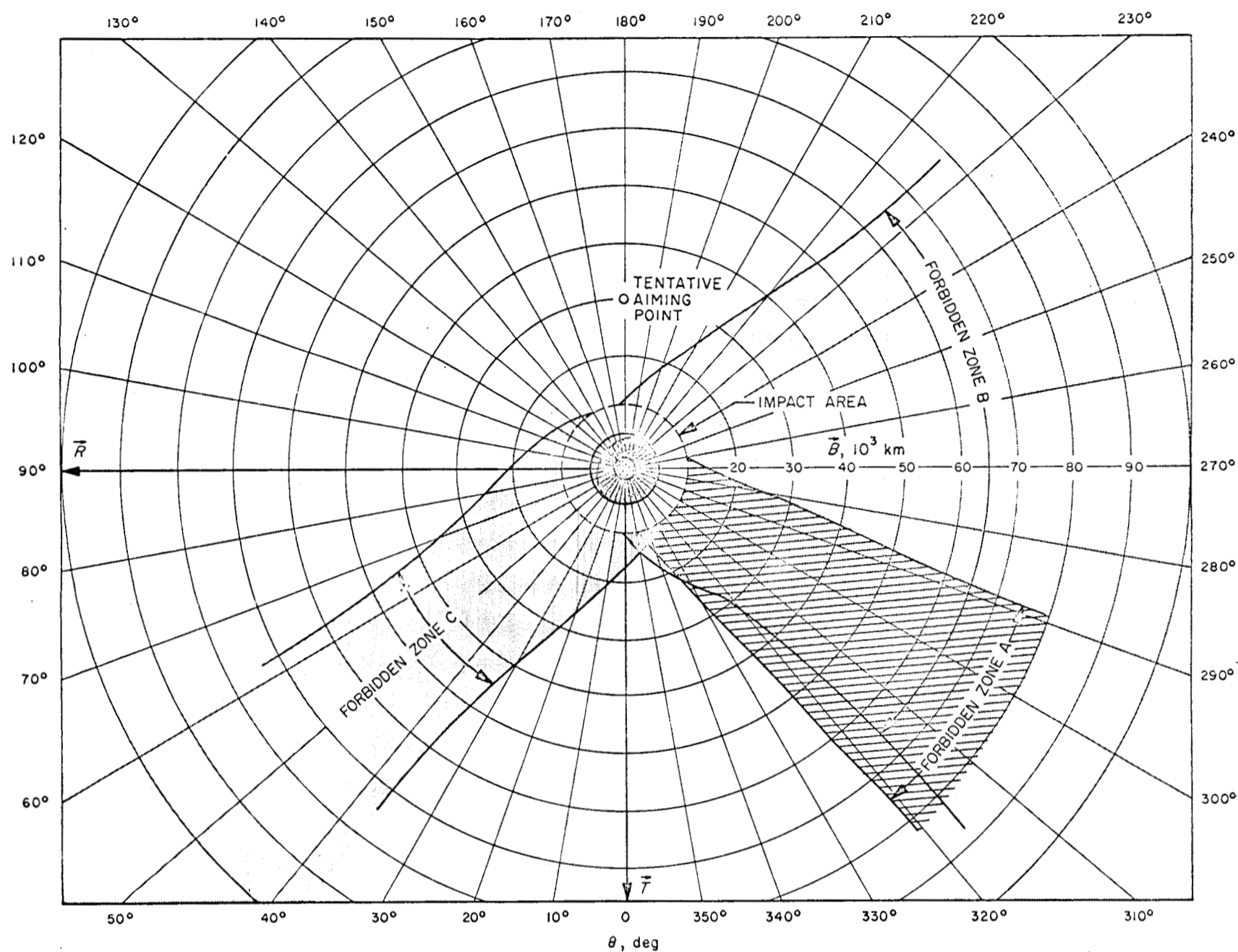


Fig. 8. Aiming-point diagram

2. Potential ambiguities associated with two-degree-of-freedom tracking systems

Since there is a limited amount of time available for the investigation of potential configurations, techniques have been evolved by which the look-angle problem for a proposed configuration can be analyzed in a systematic and straightforward manner. One technique which enables such analysis in relationship to aiming-point limitations is the shadowgraph technique, as follows:

1. For each looking element required, a shadowgraph is prepared which consists of the projection onto a reference diametral plane of the shadow cast by the spacecraft from the

looking element on a celestial sphere. There are actually two such two-dimensional shadowgraphs for each element, one for each half of the celestial sphere.

2. For a selected orientation of the spacecraft reference plane to the flight path, the target track on the celestial sphere is superimposed on the shadowgraphs, and the resulting diagram is examined for interferences.
3. Effects of vehicle reorientation with respect to the flight path can be examined in terms of corresponding target tracks.

4. By simultaneously examining several shadowgraphs for different looking elements, the several look-angle problems can be analyzed simultaneously.

To illustrate this process, we consider a hypothetical Mariner spacecraft, which is intended to perform a flyby mission to Venus, and which includes an articulated planet-seeker mounting various sensing instruments. The planet-seeker is designed to maintain a planet-centered orientation during the entire flyby maneuver. It is assumed that the spacecraft attitude remains fixed in relationship to the Sun and the star Canopus during the maneuver. Such a spacecraft is illustrated in Fig. 9.

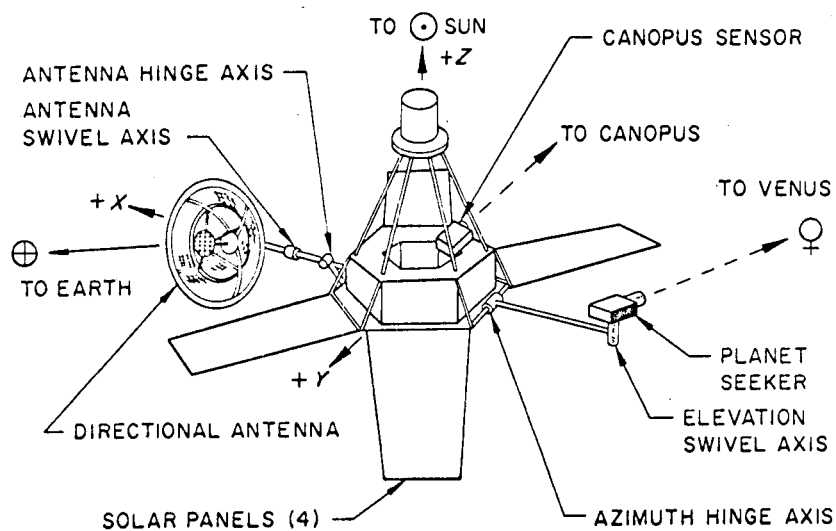


Fig. 9. Hypothetical planetary spacecraft configuration

The spacecraft shadow, as viewed from the planet-seeker, is illustrated in Fig. 10. When this shadow is projected onto the reference diametral plane, we obtain two-dimensional shadow diagrams conveniently graphed in polar coordinates. One-half of such a shadow diagram is illustrated in Fig. 11 (see next page). With a Sun-oriented spacecraft, as in this case, it is usually convenient to choose a diametral plane normal to the Sun line. Roll orientation with respect to this reference plane is more or less arbitrarily established in relationship to spacecraft geometry.

To define the target-tracking parameters, we establish a spacecraft-centered coordinate system involving what we call cone and clock

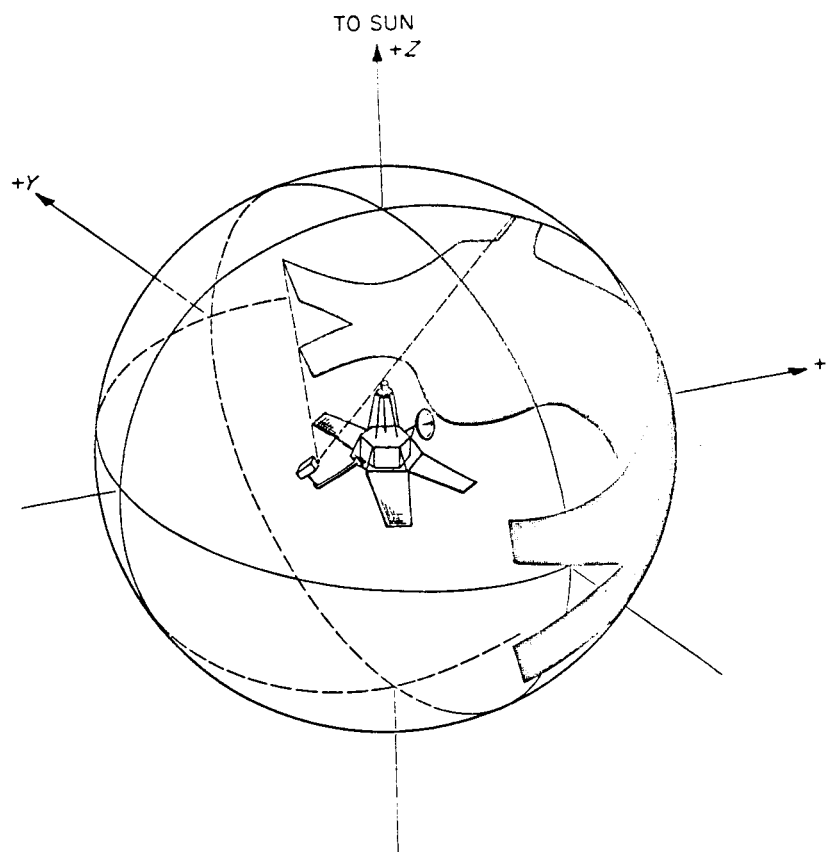


Fig. 10. Spacecraft shadow seen from planet-seeker

angles. This system is illustrated in Fig. 12. The cone angle is defined as the angle between the Sun line and the target line. The clock angle is measured in the plane normal to the Sun line, and

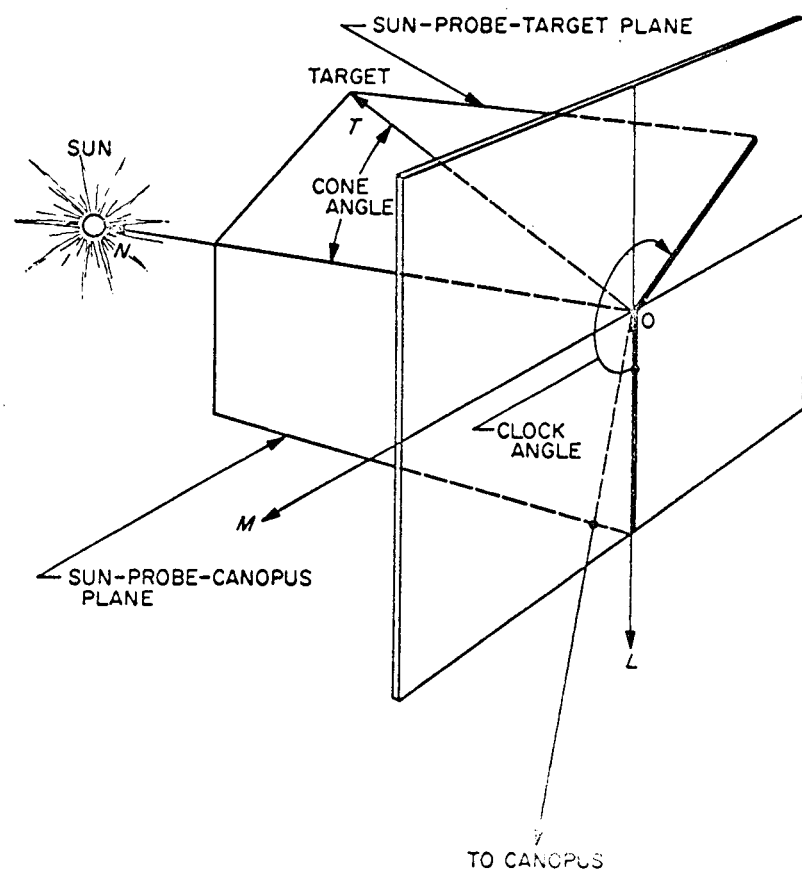


Fig. 12. Cone-clock coordinate system for spacecraft

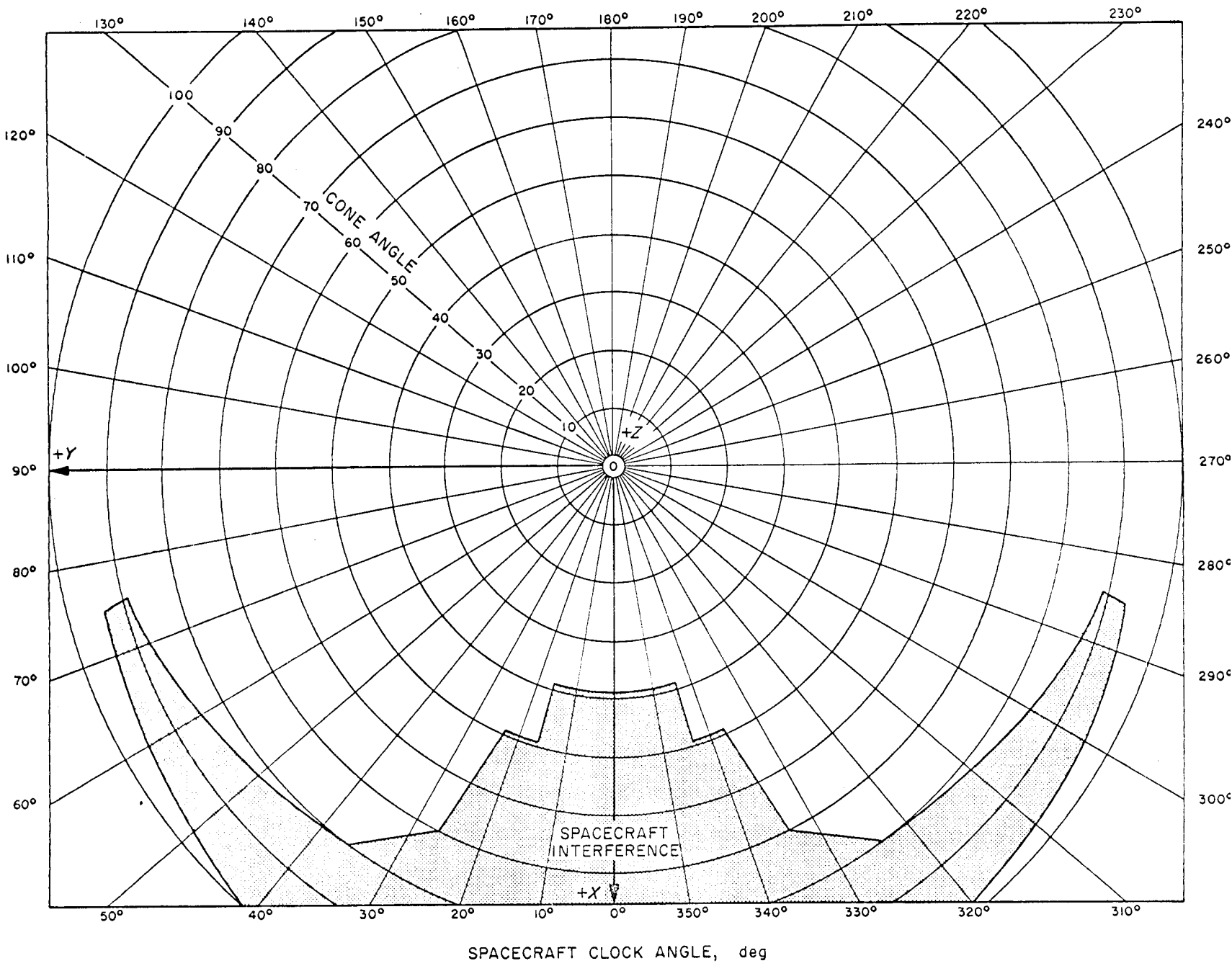


Fig. 11. Two-dimensional shadowgraph

is referenced to the Sun-spacecraft-Canopus plane. Thus, we obtain a polar coordinate system similar to that used for the shadowgraphs. The cone angles are identical for the two systems, and since the plane in which clock angles are measured is the same as the reference plane chosen for the shadowgraphs, the relationship between the clock angles is determined by the positioning of the Canopus sensor on the spacecraft.

For a selected Canopus-sensor placement, a typical target planet track for one selected aiming point is illustrated in Fig. 13 and 14. It is to be noted that the initial target direction is independent of the specific aiming point selected for the near flyby. This is because the differences in specific aiming points are very small in comparison with the overall geometry of the solar system, which establishes gross aspects of the interplanetary trajectory. Different selected aiming points will therefore correspond to different tracks, originating at a common point. Although certain geometrical relationships do prevail, the same is not true of the receding asymptote or terminal point.

We now superimpose the spacecraft shadowgraphs on the planet track diagrams and examine the resulting composite diagrams for interference. One-half of such a pair is illustrated in Fig. 15. It is to be noted that for the spacecraft-Canopus orientation selected, the target path crosses over a region of spacecraft interference, indicating an unacceptable arrangement. By simply rotating the planet track graph with respect to the spacecraft shadowgraph as shown in Fig. 16, the effect of moving the Canopus sensor can be determined. In an actual case, of course, there would be several shadowgraphs for different items of equipment, and each pair of diagrams would have to be rotated correspondingly in order to examine the overall situation.

To illustrate the concept of the shadowgraph technique, the simplifying assumptions have been made in the previous discussion that the target consists of a point, and that the looking element sights along a line. Neither of these assumptions is valid for the real case. For

example, the Mariner 2 passed approximately 22,000 miles from the surface of Venus. At that distance the planet subtended an angle of approximately 23 degrees and the target track was actually a target band. Additionally, almost all sensors have a field of view describable in terms of an included angle. This is taken into account by establishing an effective shadow area outside of which there is no encroachment into the sensor field. The shadowing effect may be of minor importance very close to the shadow boundary. A typical case which superimposes the target envelope and the effective spacecraft shadow is illustrated in Fig. 17.

As mentioned earlier, there will be different tracks in the cone-clock coordinate system for different aiming points, and for a particular spacecraft configuration, it may be that certain aiming points are acceptable, and others not. Accordingly, for a given spacecraft orientation (as established by location of the Canopus sensor) we can check the target tracks relative to the spacecraft shadow. The superposition of several such target tracks is shown in Fig. 18. Here, for purposes of simplicity, the target is again represented as a point. For the case illustrated, it can be seen that only those aiming points which are in a certain region will be acceptable from the tracking standpoint, and that, as mentioned earlier, the initial target corresponding to the incoming asymptote is the same for all tracks. Information obtained from such an analysis can be readily summarized on an aiming-point chart which now incorporates all of the forbidden zones, including those for which the target cannot be kept in continuous view of the tracking sensors throughout the flyby pass. Such a chart is illustrated in Fig. 19, and can be used to assess what is practicable with a proposed configuration in relationship to scientific requirements for the mission.

The basic technique described in this section can be applied in the design process in connection with the somewhat more subtle and tricky problems associated with selection of hinge axes for articulated members, and analysis of servo requirements. Only the relatively simple problem

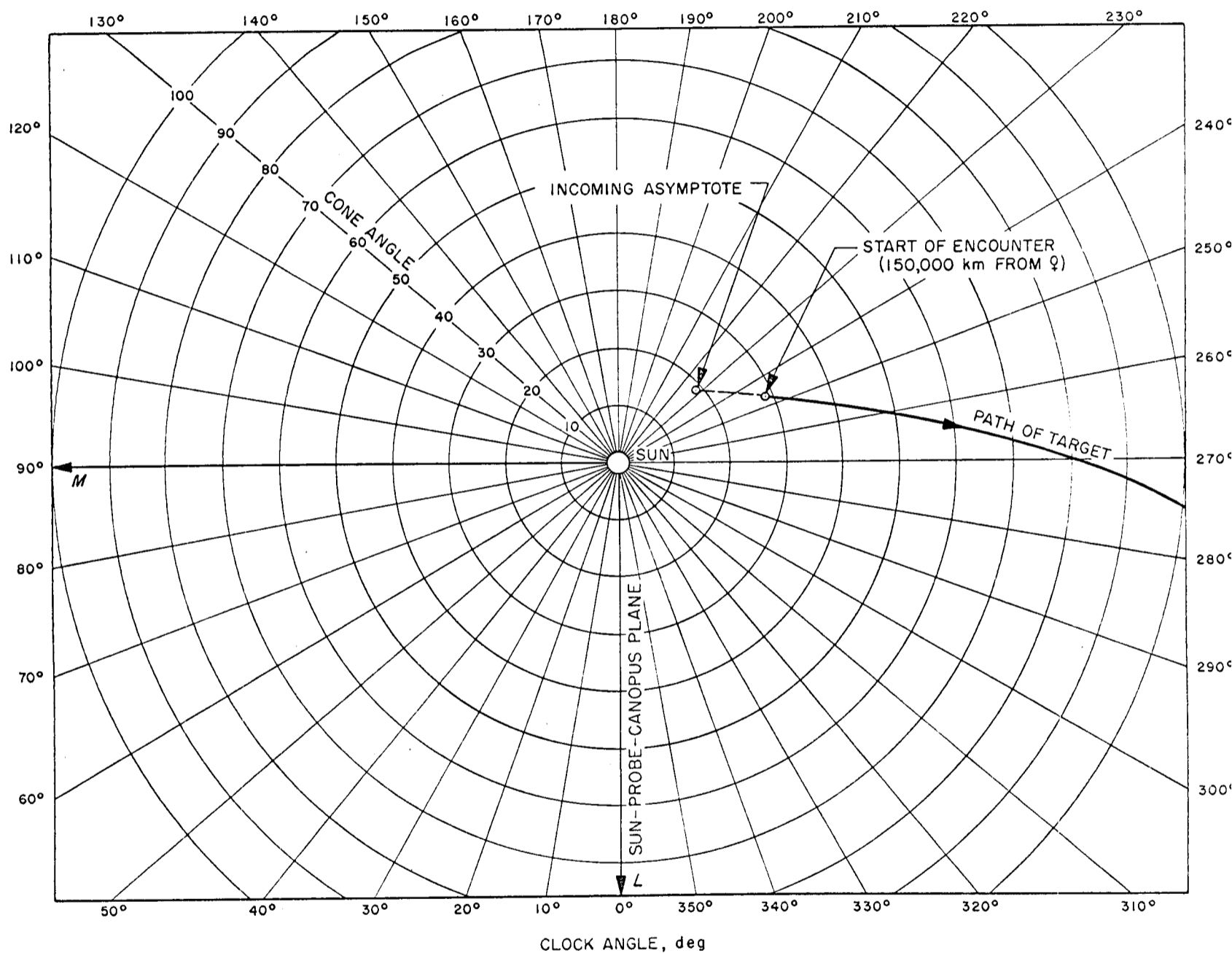


Fig. 13. Target track in cone-clock coordinates

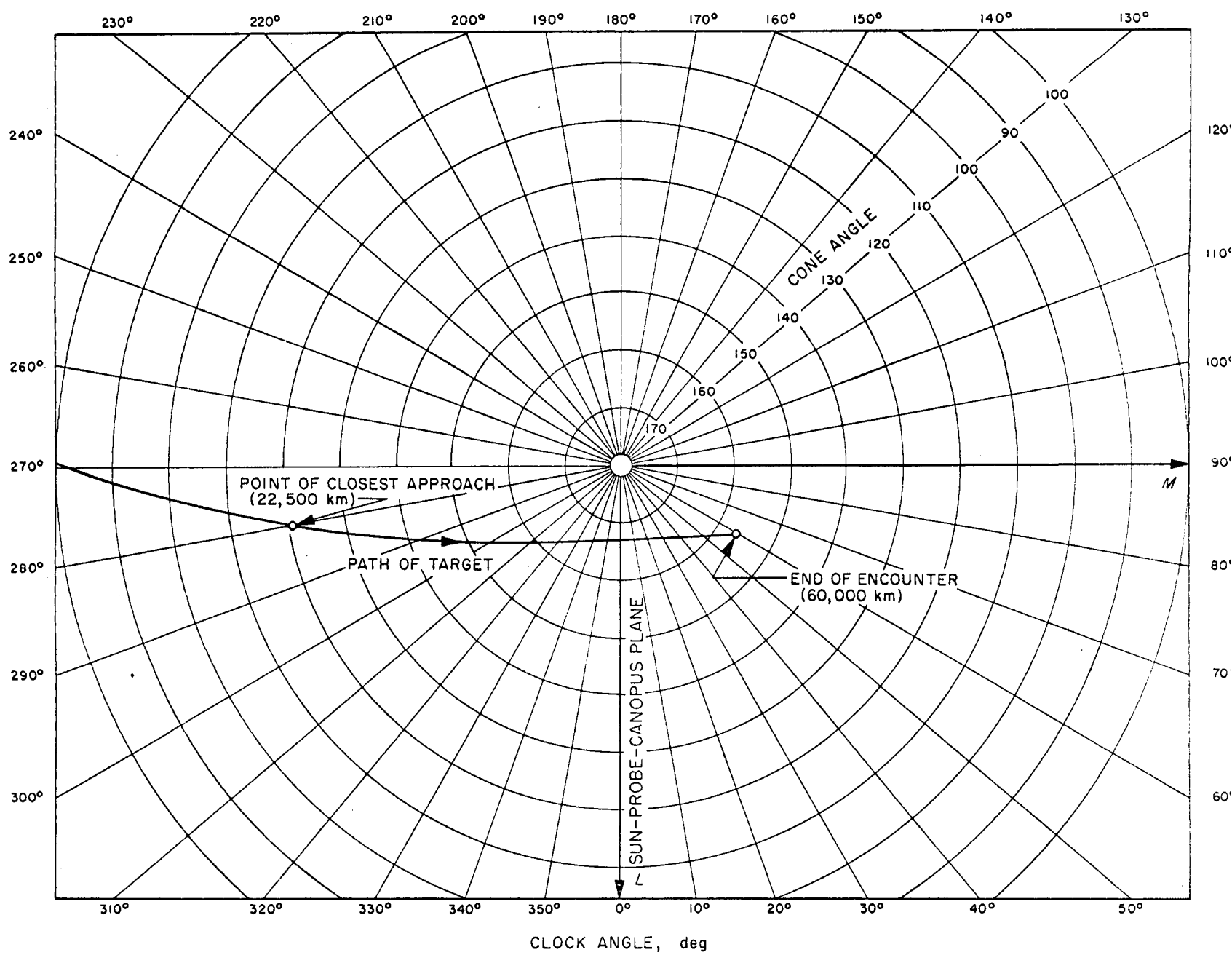


Fig. 14. Target track in cone-clock coordinates

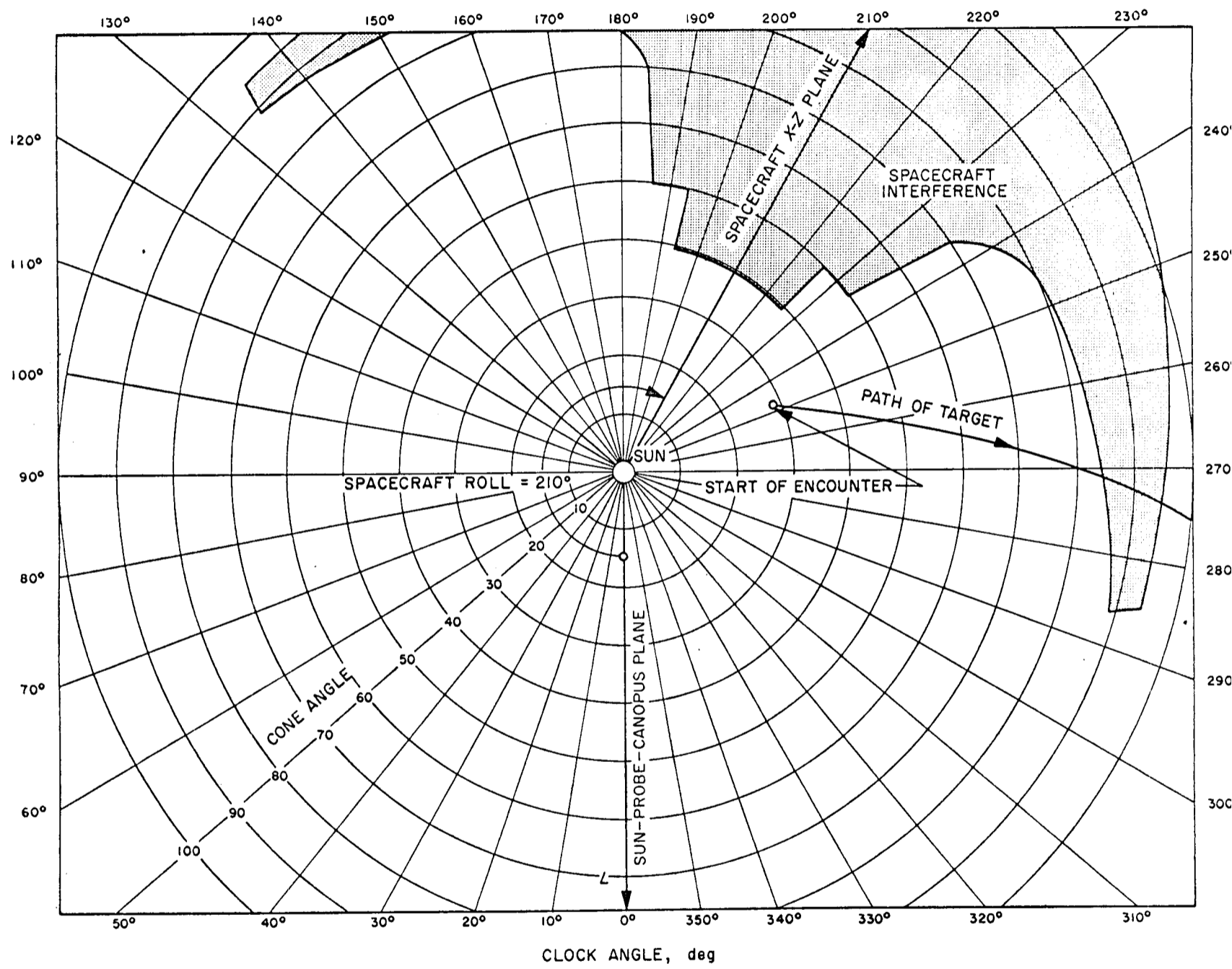


Fig. 15. Target track superimposed on shadowgraph--showing interference

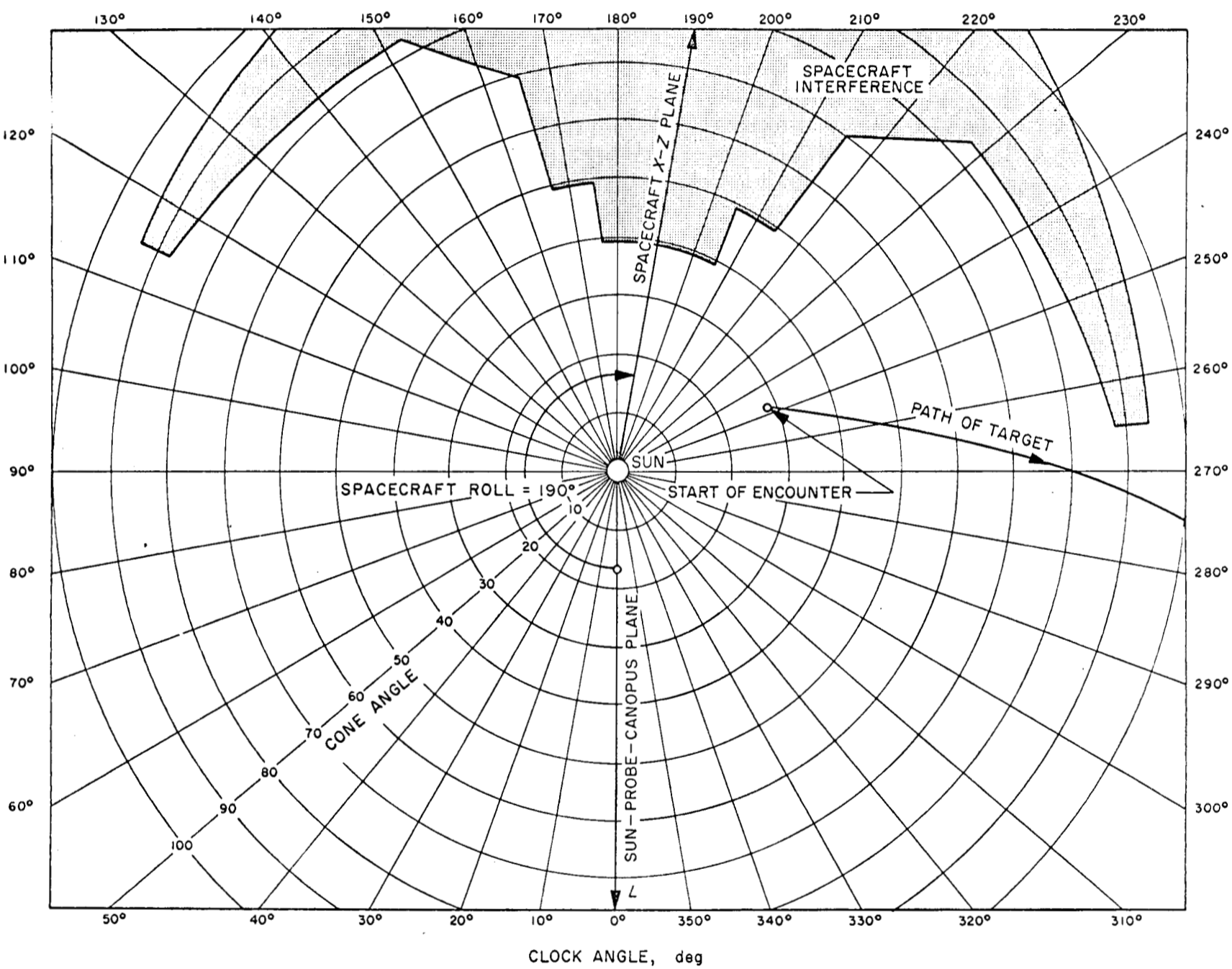


Fig. 16. Target track superimposed on shadowgraph--interference eliminated

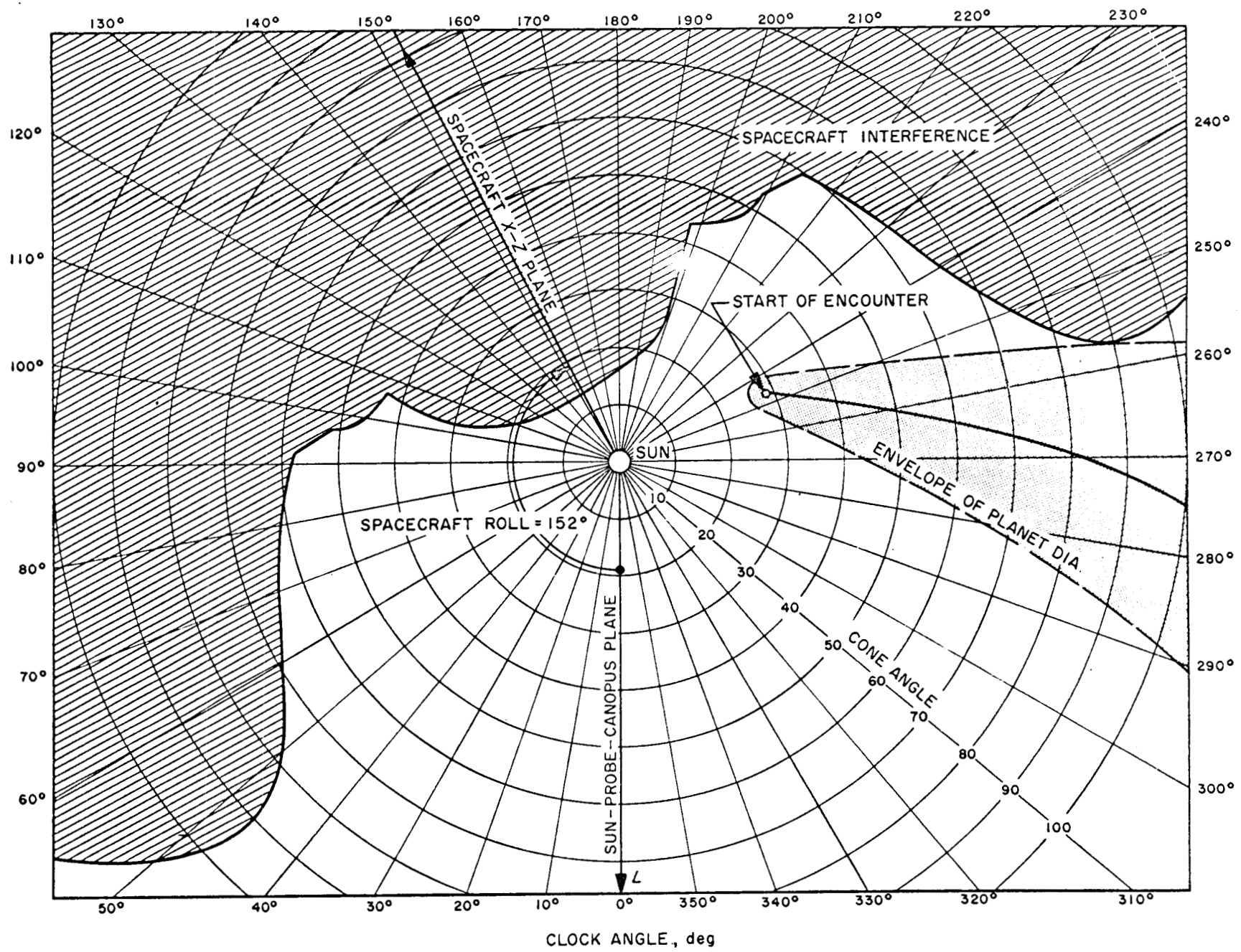


Fig. 17. Target envelope superimposed on effective shadowgraph

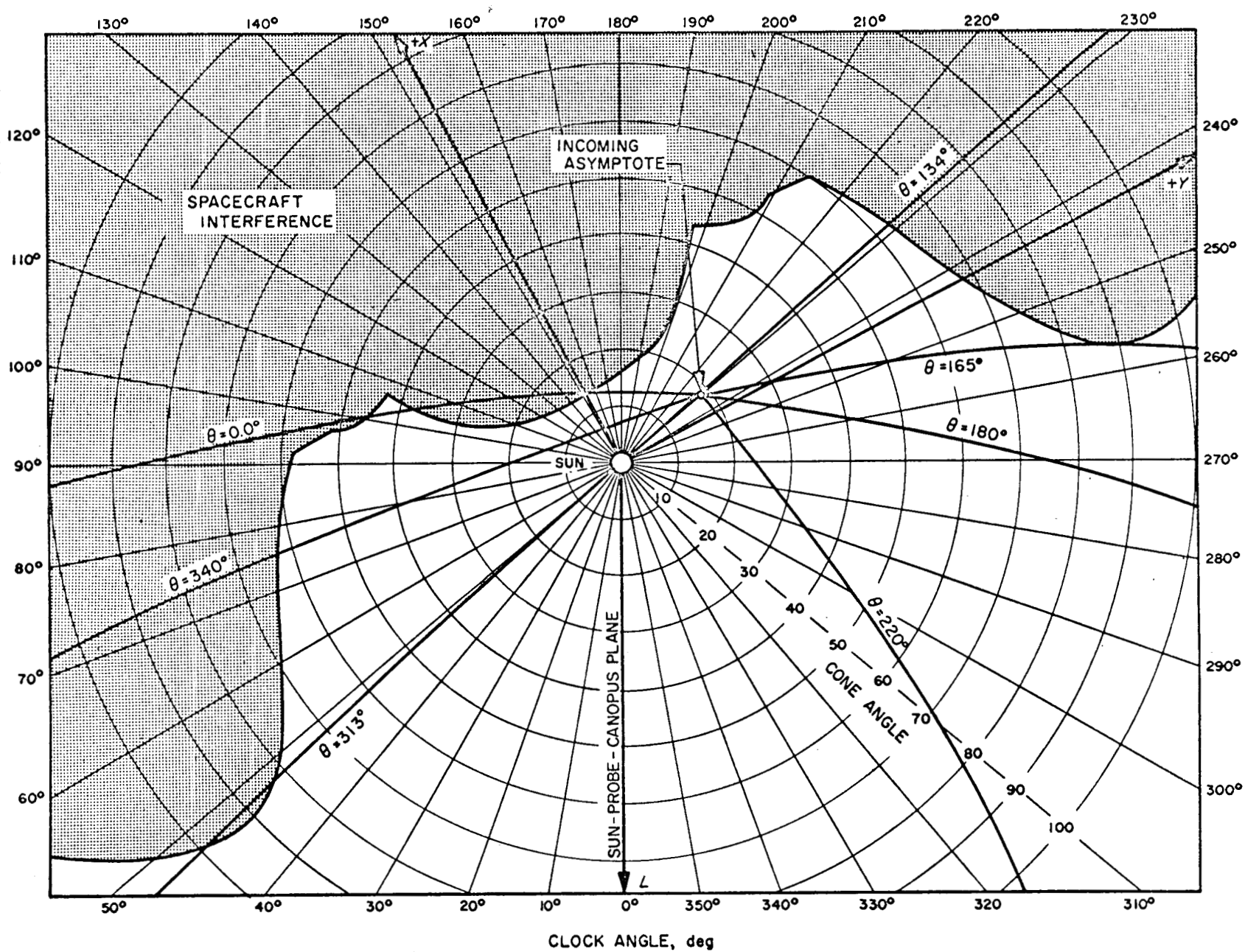


Fig. 18. Target tracks illustrated for several aiming points

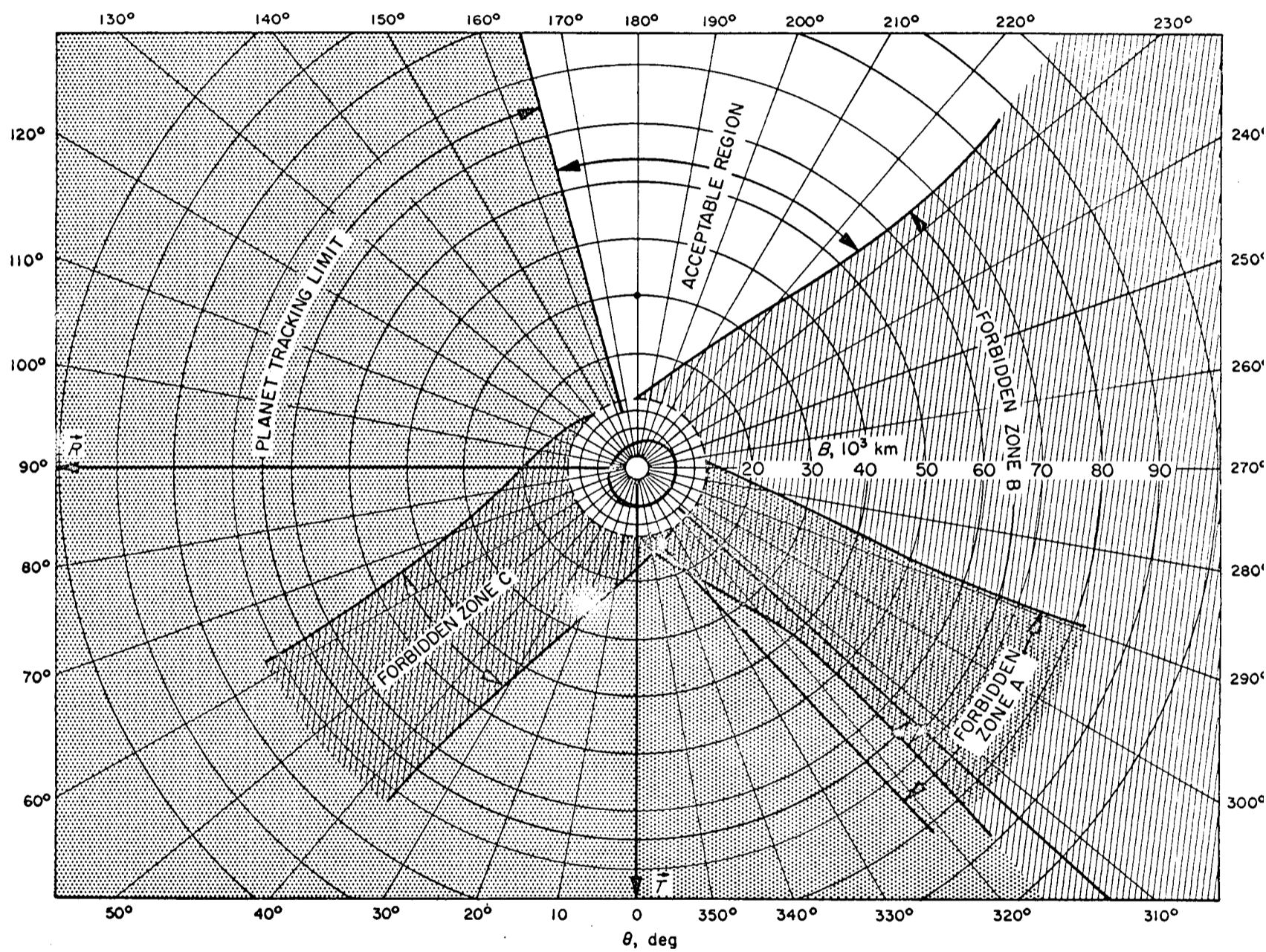


Fig. 19. Aiming-point selection chart for typical mission

of basic target-tracking has been considered here. Again, it must be recognized that the preceding discussion has dealt only with that portion of the flight involving target encounter. Other phases of the flight, including the immediate postinjection phase and the midcourse-maneuver phase, must also be investigated and can involve critical problems. In fact, the cruising portion of flight may also involve difficulties associated with gimbal lock, or articulation limits for the spacecraft antenna.

In any case, the method described is a tool only. It will not produce a design. Only the properly applied imagination of the engineering team can generate a satisfactory configuration. However, since the geometrical problems are both complex and difficult to visualize, tools such as the shadowgraph technique can very significantly improve the efficiency with which an acceptable design is evolved.

SPACECRAFT STRUCTURAL DESIGN

Requirements on Structure

With few exceptions, essentially every piece of material aboard a spacecraft serves some structural function, and is shaped or sized or processed with some consideration for its structural properties. However, a relatively small fraction of the total spacecraft weight is devoted to connective structure in the classical sense. The fraction varies between approximately 10 and 25 percent, depending upon mission requirements and the degree of sophistication permitted in design.

The connective structure of a spacecraft is designed to a set of requirements very similar to that imposed for any structural system. The only fundamental differences between a spacecraft structure and that for an office building, say, result from differences in the relative emphasis placed upon the several requirements. Moreover, it has been our experience that the design approaches and methods of analysis are really quite similar to those now being applied by structural engineers working in other fields of

application. A number of our most competent structural engineers have been trained as civil engineers.

Basic requirements which the spacecraft structure must meet can be briefly outlined as follows:

1. The structure must be compatible with the geometric configuration required for the spacecraft. This includes maintenance of all critical alignments between separate items of equipment.
2. The structure must be of adequate strength to withstand loads imposed during handling, launch, and flight.
3. The structure must have acceptable dynamic characteristics.
4. Like every other spacecraft system, the structure must have minimum weight.

Aside from the emphasis placed upon minimizing structural weight, perhaps the most significant respect in which the spacecraft structural problem differs from that encountered in many engineering fields results from the relatively minor significance of the static or quasi-static loads. With few exceptions, it can be stated that dynamic considerations, including associated stresses, design the spacecraft structure. This being the case, it is indeed unfortunate that the dynamic environment within which the structure must perform is not yet well defined or fully understood. A great deal of effort is currently being devoted to improving this situation and attempting to establish substantially more rational criteria for design. In this section, I will primarily describe the approaches and criteria we are now using with some evaluating comments.

A basic fact of life with many of our current spacecraft is that the structure is almost entirely designed to meet conditions encountered during the launch phase of flight. All loads of structural significance are introduced into the spacecraft through the launch vehicle connective

structure. This leads to an interesting predicament:

1. From the viewpoint of a launch vehicle the spacecraft can be thought of as an insignificant "black box." Elastic characteristics of the spacecraft are of little significance to the launch vehicle structure.
2. From our point of view the spacecraft is an elastic structure. It can receive and interact with only those load inputs which the local launch vehicle structure is capable of introducing.

Perhaps the most rational approach to spacecraft design would be to consider it as one piece (an extension) of the total launch vehicle. Unfortunately, until very recently, it has not appeared that organizational arrangements or analytical tools would permit the design to be developed from this point of view. As a consequence, we have generally been designing to fairly typical "black box" vibration specifications expressed in terms of qualification test requirements. These requirements have been established, in most instances, on the basis of extrapolated observations of dynamic conditions encountered on launch vehicles during previous flights. Almost inevitably, the data have been obtained on different launch vehicles, or with different payloads than those of current design, and their specific interpretation is anything but straightforward.

A typical qualification test specification as currently employed includes the following provisions for forced-vibration input at the spacecraft-to-launch-vehicle joint:

1. Longitudinal: Sinusoidal vibration sweep from 5 to 100 cycles per second with acceleration amplitude of 2 g's rms.
2. Lateral (each axis): Sinusoidal vibration sweep from 5 to 100 cycles per second with acceleration amplitude of 1 g rms.

The sweep rate is increased linearly with frequency and the time required for the entire sweep is specified at approximately 6 minutes.

For the types of spacecraft currently under development, implications of the above test requirement largely design the structure.

Environment and Loads

The major sources of loads on a launching rocket are illustrated schematically, and with a great deal of artistic license, in Fig. 20. Loads on the launching rocket are of significance to the spacecraft primarily because it is the transmission of, and reaction to, these loads by the launch vehicle which establish the load environment of a spacecraft. Generally, these loads fall into two classes: those generated within the launching rocket, and those introduced by the atmosphere. Both airborne noise and atmospheric buffeting fall into the latter class.

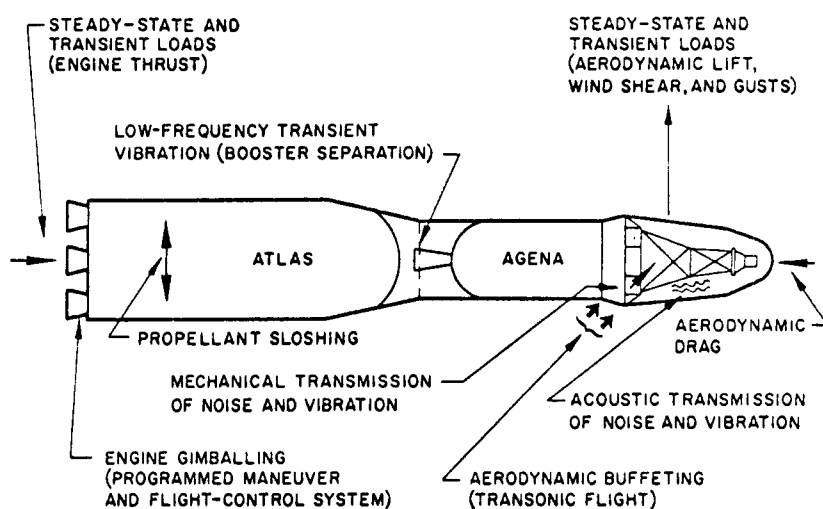


Fig. 20. Schematic representation of loads on typical launch vehicle

Static and quasi-static axial and horizontal loads introduced into the spacecraft in a typical case are illustrated in Fig. 21. Here we see that the axial acceleration provided by rocket motor thrust rises as fuel is consumed and until rocket cutoff. The series of steps corresponds to vehicle staging. It is to be noted that the maximum acceleration does not normally exceed approximately 6 to 8 g's. Side loads can only be indicated by an envelope, since quasi-static side

loads are usually the result of launch vehicle maneuvers compensating for atmospheric gusts.

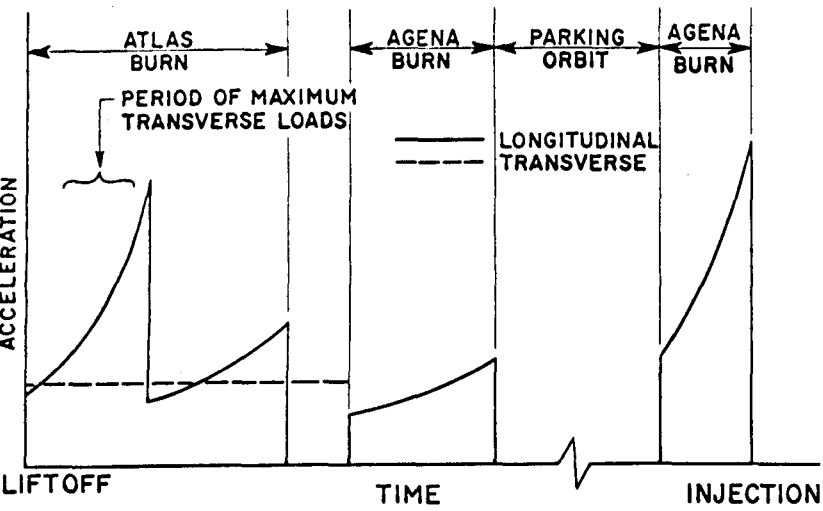


Fig. 21. Typical static and quasi-static loads on spacecraft during launch

In the category of dynamic load inputs from the launch vehicle, we have random noise originating primarily in rocket motor combustion and in aerodynamic buffeting. The typical energy involved in this wide-band vibration is illustrated in Fig. 22. The first peak occurs very early in the launch sequence, and originates in the rocket motor. The second peak occurs when transonic velocities through the atmosphere are encountered, and is associated with aerodynamic buffeting, primarily around the shroud base. This type of noise-loading is of much greater importance to the electronic and electromechanical equipment than to the structure.

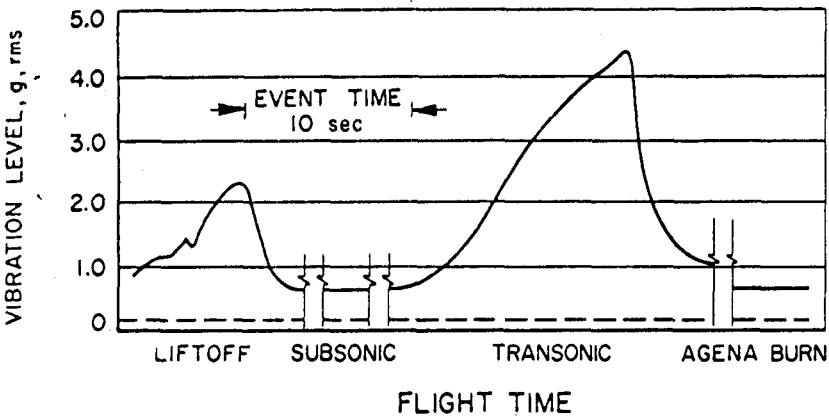


Fig. 22. Typical vibration-energy environment of spacecraft during launch

For the range of frequencies which are of primary interest from the spacecraft structural standpoint, dynamic load inputs are associated

primarily with the bending-mode response of the launch vehicle to disturbances. Bending-mode shapes for a typical booster are illustrated in Fig. 23 for the first three bending modes. The mode shapes have been normalized to the tip deflection. We see that the forced motion of the spacecraft can be described in terms of a translation and corresponding rotation, or alternatively, in terms of rotation about an effective center corresponding approximately to the nearest node point. It is to be noted that the mode shapes will change significantly as fuel is consumed. Characteristics illustrated are for a particular time of flight. Bending frequencies also change during flight as fuel is consumed.

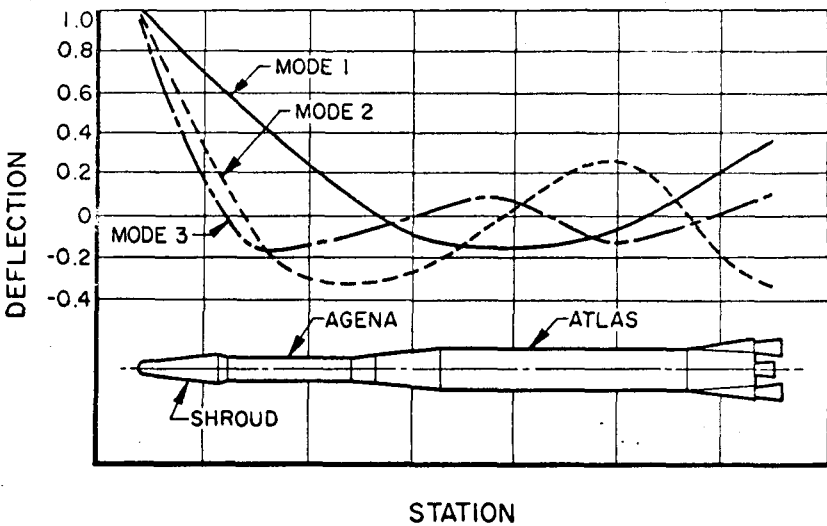


Fig. 23. Typical booster bending modes

In addition to loads associated with white noise and with booster bending modes, the spacecraft will experience transient load inputs associated with specific events, such as engine cutoff or staging. Figure 24 is a telemetered record

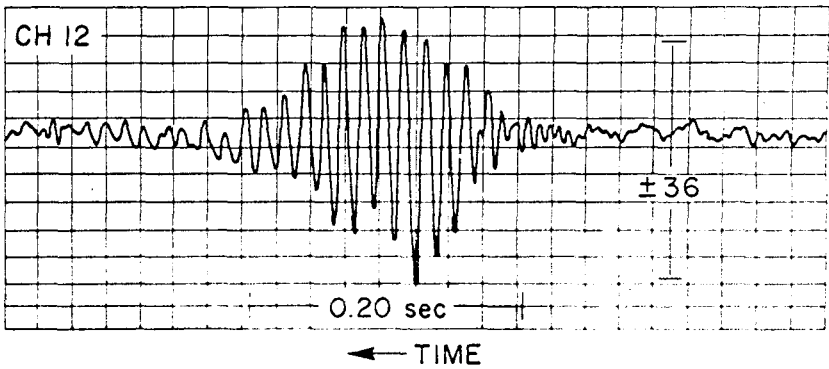


Fig. 24. Typical transient vibration condition

obtained from an accelerometer located on the launch vehicle structure near the spacecraft interface during engine cutoff. It is to be noted that the accelerometer recorded a specific frequency for a number of cycles. Such a transient load situation may or may not be significant to the design.

One special structural problem deserves mention. This is the problem of atmospheric entry vehicles, where very large loads are encountered associated with deceleration of certain spacecraft as they enter the atmosphere of a target planet. For unmanned spacecraft, peak deceleration values of approximately 150 g's are likely to be encountered, and the problem is greatly complicated by the very large heating rates as kinetic energy is dissipated through atmospheric drag.

In summary, we have static or quasi-static loads and dynamic loads, the latter of either transient or long duration. The dynamic loads are defined in terms of a qualification test specification applicable to the entire spacecraft, including the structure, and assert a major influence over design.

General Approach to Design

There is nothing particularly unusual about the design approach used for spacecraft structures. The approach is iterative in character and proceeds in parallel with other activities essential to spacecraft development, such as configuration and temperature control development. Frequently the structural design concept evolves from a previous spacecraft or project. The first step in the process is to establish a structural configuration concept. The next step, traditional in the aircraft industry from which space technology has largely derived, is to break the total structure down into substructures which can be decoupled from each other. The Mariner 2 spacecraft, as illustrated in Fig. 25, can be considered as a typical case. Here the structure has been broken down into the main equipment enclosure (or "hex"), a superstructure, the solar panels, and the high-gain antenna. Each of these substructures could be analyzed and initially treated more or less independently of the whole.

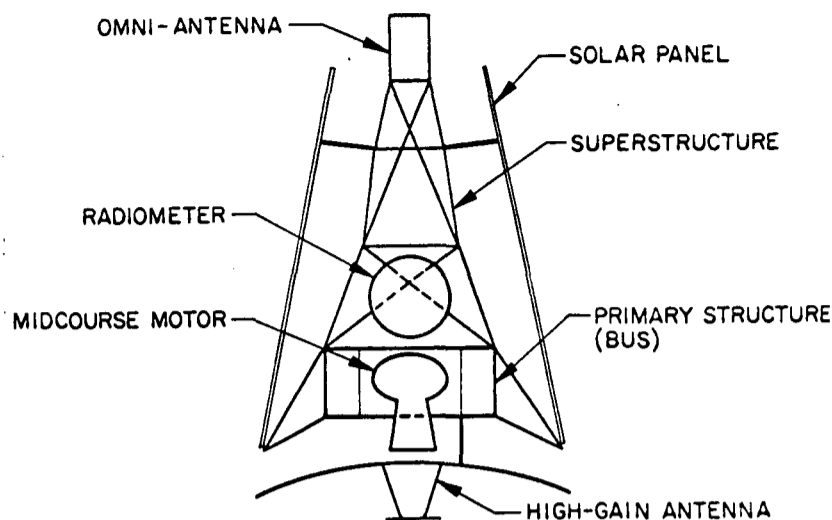


Fig. 25. Schematic of Mariner 2 structure

The several substructures are then designed and analyzed for the loads imposed, including the connective loads between substructures. Commonly, although this occurs almost automatically, it is desirable that the natural frequencies for the several substructures be decoupled from each other. This simply means that no lightweight substructure is forced to serve as a tuned, lightly damped vibration-absorber.

Finally, an overall analysis of the assembled substructures, constituting the spacecraft structural system, is carried out. This analysis uses generalized masses and generalized stiffnesses for the first few modes of each of the substructures, up through approximately 100 cycles per second. This procedure, working from substructures through the overall structure, is repeated as many times as required to produce an acceptable design.

During the process previously outlined, considerable early use is made of highly simplified models to test gross aspects of structural concepts. In addition, of course, analytical methods are applied extensively, using high-speed digital computers, such as the IBM 7094 computer. The spacecraft is ultimately subjected to qualification tests using static-loading systems and large shakers.

Analytical Methods

Aside from ordinary stress analysis of detailed members, the primary analytical tool is



a set of computer programs based upon stiffness matrices for the spacecraft structures. The stiffness matrix approach has been selected in preference to other analysis methods, primarily because the input is in a simple form, it is adaptable to any type of framework, and it provides a complete analysis.

The program which is used generates the stiffness matrix for a particular type of structure from geometrical data, and performs static and normal mode analyses by simultaneously solving the following stiffness and inertial matrix equations:

$$U = K^{-1}F$$

$$\frac{1}{\omega^2} U = K^{-1}MU$$

where

F = a matrix of static loads

M = a matrix of inertia terms

U = a matrix of static deflections

ω = the circular frequency of a normal mode

The stiffness equation simply expresses in matrix form the fact that at any joint in a structure a component of load applied to the joint must be in equilibrium with member stresses reacting on the joint in the same direction, and that in a linear structure such member stresses are proportional to member deflections. The second matrix equation relates the inertial forces for free vibration in a normal mode to deflection stresses. Masses are assumed to be lumped at the joints. With the geometries, elastic properties, inertial properties, and static loads given for all joints in the system, simultaneous solution of these matrix equations will permit determination of (1) deflections and member stresses for static loadings, and (2) frequencies, mode shapes, and member stresses during free vibration in normal modes.

Computations are carried out using a high-speed digital computer, and preparation of material for the computational program is accomplished in a routine and simple fashion. Specific

programs have been prepared for each of the following four types of structure:

1. Three-dimensional structure, pinned joints
2. Three-dimensional structure, rigid joints, circular-member cross sections
3. Planar grid structure, rigid joints, loaded in-plane
4. Planar grid structure, rigid joints, loaded normal-to-plane

Each program is capable of handling a maximum of 130 degrees of freedom.

Figure 26 illustrates a typical "space-frame" structure, in this case the superstructure for the Mariner 2 spacecraft. This structure was treated as a three-dimensional structure with pinned joints, as shown in Fig. 27. Good agreement was obtained between the results of this analysis and subsequent structural tests.

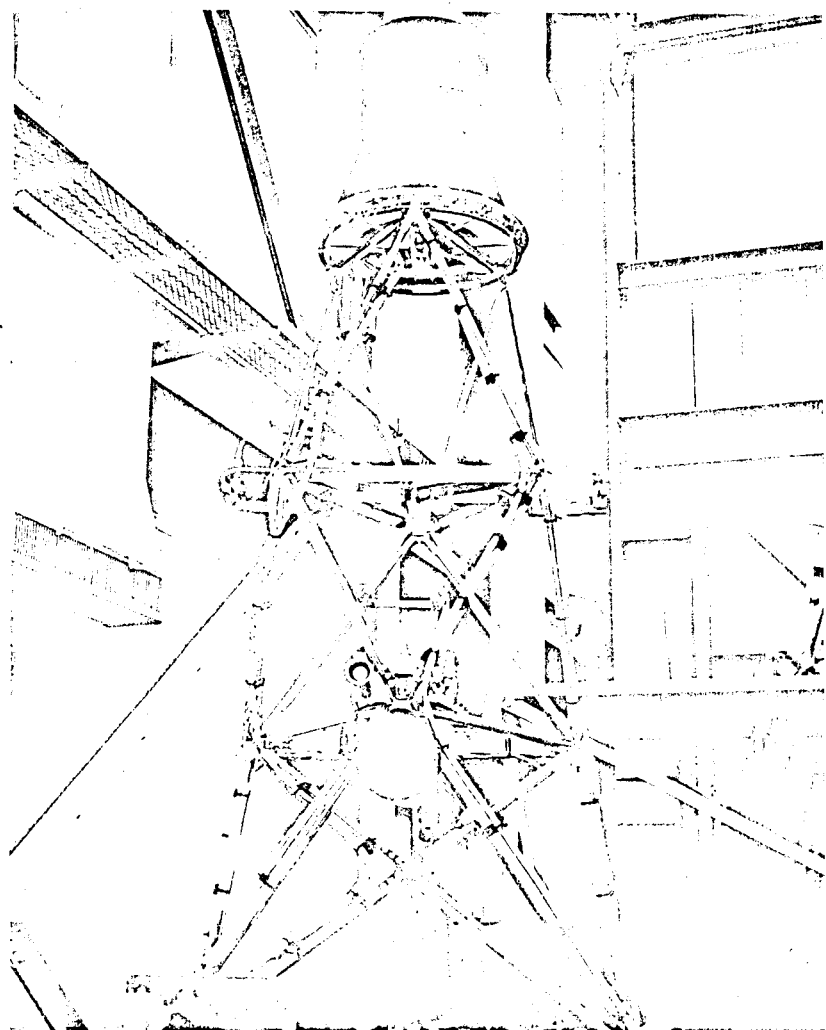


Fig. 26. Mariner 2 superstructure

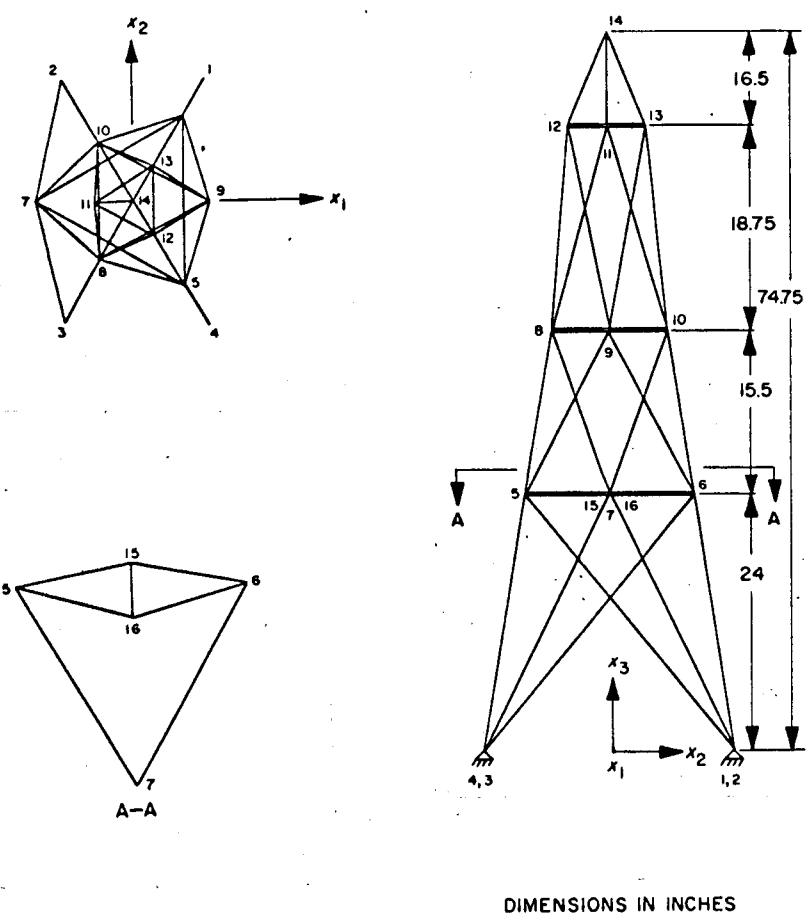


Fig. 27. Schematic representation of Mariner 2 superstructure for analysis

A different situation is represented by the Mariner 2 solar panels, shown in Fig. 28. These solar panels consisted of very light, stiffened plates restrained at six points. Since the loads and motions normal to the solar panel plane were of primary interest, the solar panel was idealized as illustrated in Fig. 29, and the analysis program for a planar grid structure with rigid joints, loaded normal-to-plane, was applied. It is interesting to note that the technique of analyzing a continuous plate in terms of an equivalent grid structure gives quite satisfactory results. Some judgment, of course, was necessary in establishing the grid and in assigning the distributed mass to the specific joints of the grid. However, application was generally quite straightforward, and again, test results were in good agreement with the analyses.

Not all of the spacecraft structures are as readily idealized as the two previously illustrated. The main equipment enclosure, or "hex", for the Mariner 2 is illustrated in Fig. 30.

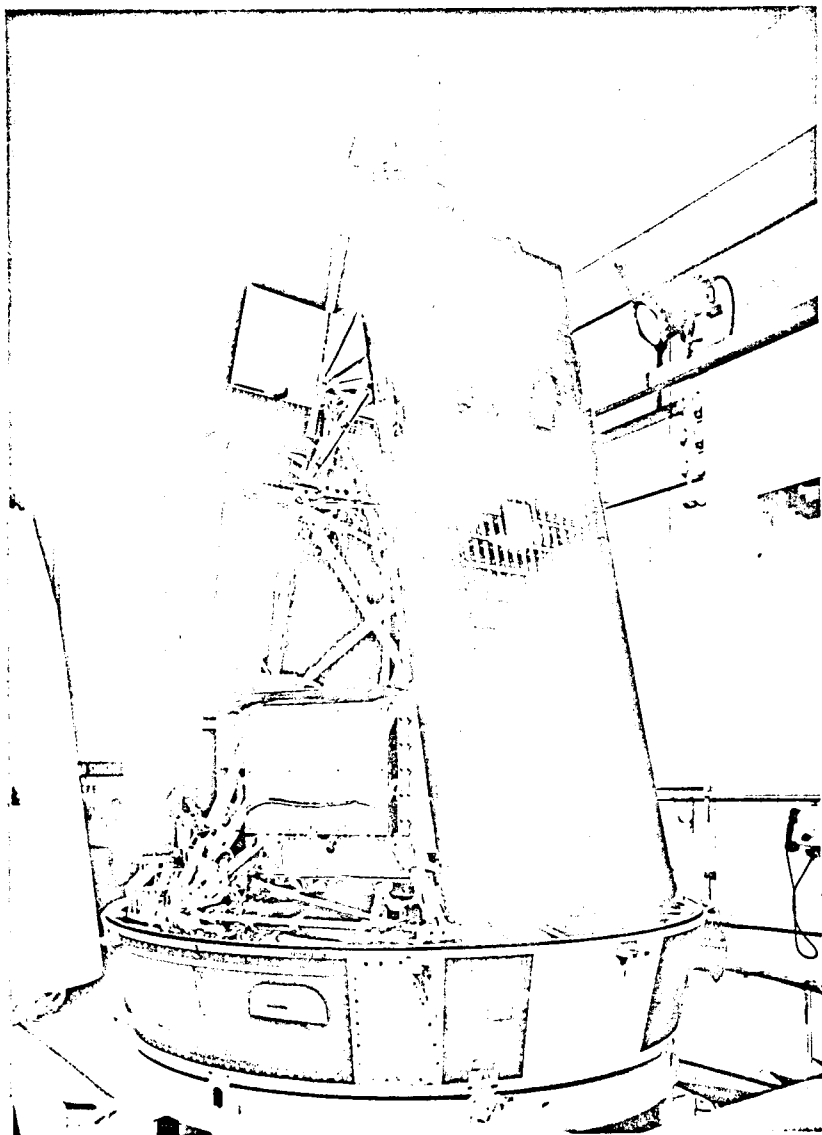


Fig. 28. Mariner 2, showing solar panels

Here the assumption of pinned joints is obviously a crude approximation, and the alternative assumption of circular-member cross sections is hardly better. Even in this case, however, results of the analysis were of considerable value in establishing some of the response characteristics and stress levels.

The technique mentioned previously for carrying out an overall analysis of the assembled substructures has only recently been introduced. The limitation of 130 degrees of freedom has heretofore constrained overall structural analyses to relatively crude approximations of the actual structure. However, the new technique largely eliminates this constraint because it enables each substructure, which has been previously analyzed, to be represented by a relatively small number of degrees of freedom associated with the modal characteristics of the



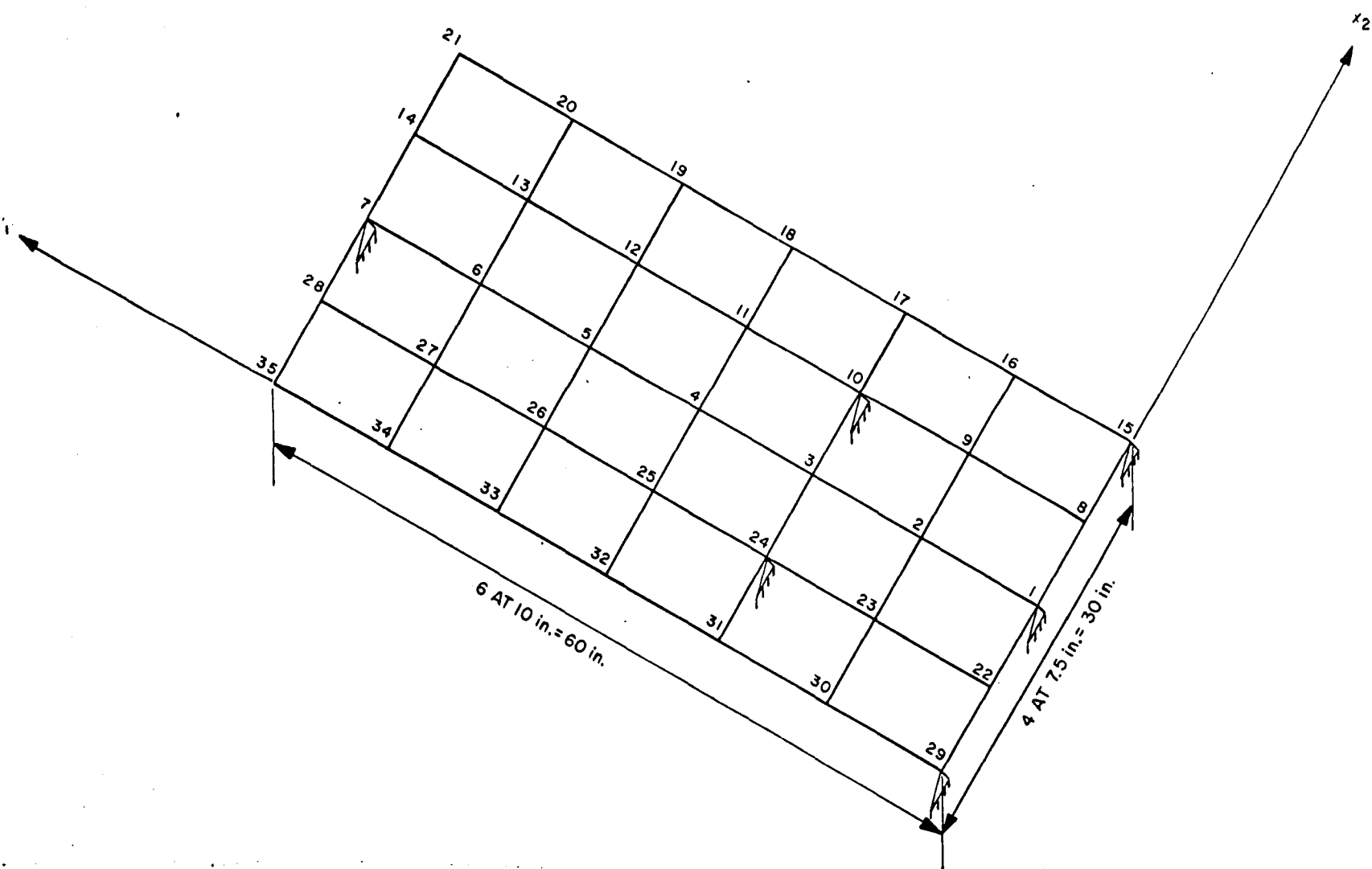


Fig. 29. Schematic representation of Mariner 2 solar panel for analysis

substructure. Thus, a particular substructure can be represented quite accurately by perhaps 20 degrees of freedom instead of the 100 or more otherwise required. This technique promises to be a powerful one in enabling overall structures to be analyzed prior to qualification testing.

Development and Qualification Tests

As noted previously, design requirements for dynamic loads are usually expressed in terms of test conditions to which the structure must be subjected. For most items of spacecraft equipment there are two types of tests: Type Approval tests and Flight Acceptance tests. The former are essentially design-verification tests, while the latter are intended to verify design execution. In the case of spacecraft structure, the Flight Acceptance tests are usually waived, it being assumed that careful inspection by various

techniques will verify the quality of fabrication applied to each individual unit.

In addition to official qualification tests, a variety of tests are performed as an integral part of the development process. Three general types of tests which are used for either or both development and design qualification are static tests, dynamic tests, and modal vibration surveys.

Except for certain specific types of flight hardware, relatively little emphasis is placed upon static tests to establish load-carrying capability of structure. However, static tests are performed in some instances, primarily to establish or confirm the elastic properties of the structure. The test setup is a conventional one employing whiffle trees, cables, pulleys, and weight pans. Our structural test laboratory is arranged so that we can build up any necessary



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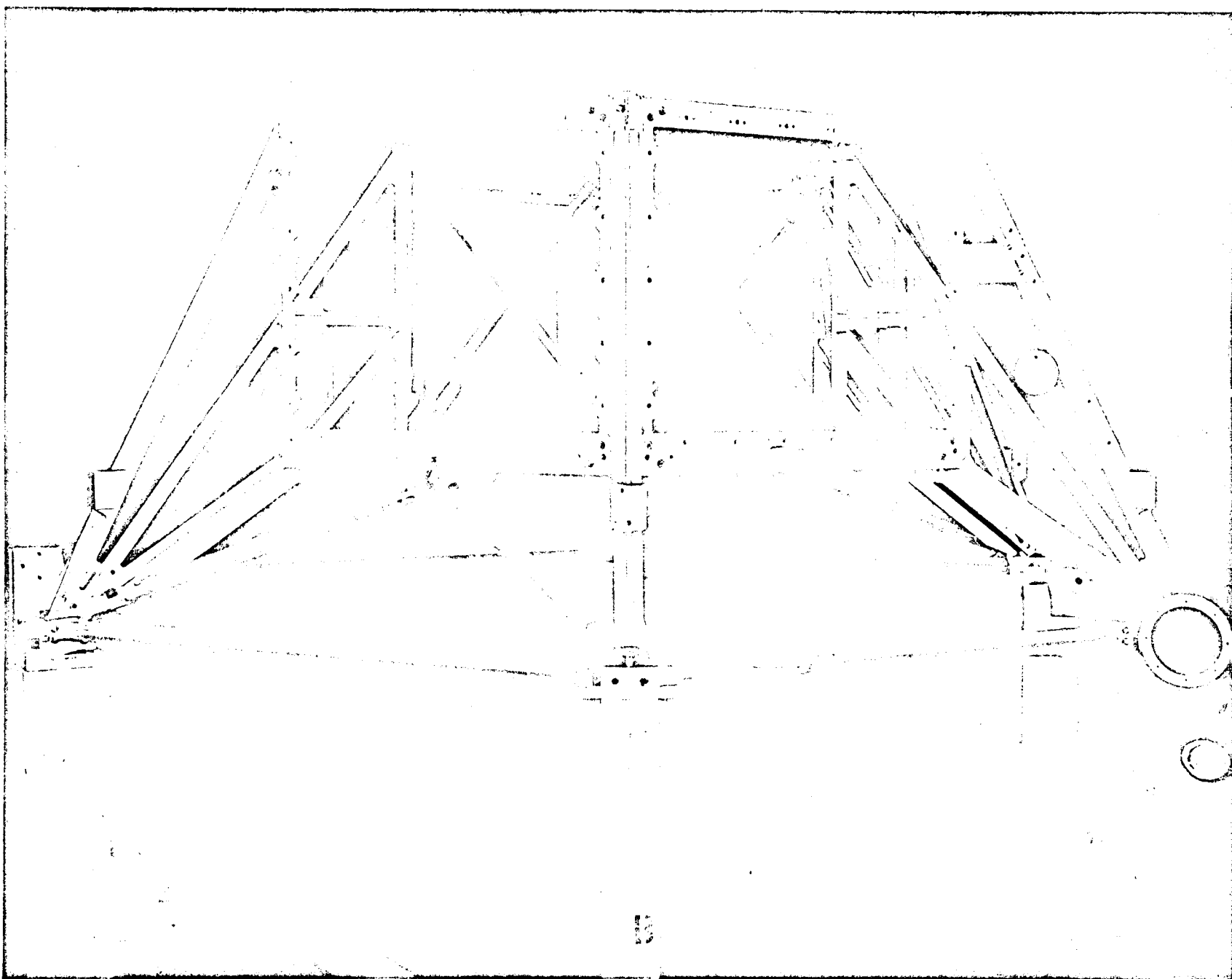


Fig. 30. Mariner 2 main structure





framework of loading fixture to meet specific test needs. Electrical strain gauges are used to determine local stress values, and deflections are generally determined by mechanical means.

With spacecraft of the sizes so far encountered in the unmanned space program we have been able to conduct dynamic qualification tests using large hydraulic and electromagnetic shakers. A typical setup for a spacecraft shake test is illustrated in Fig. 31. The structure is mounted on a special fixture designed for the purpose, and generally as stiff as possible, and supported by an oil film on a granite surface plate. A servo-controlled hydraulic shaker drives the support fixture in a horizontal plane. For a particular plane of vibration the hydraulic shaker is controlled to provide a sweep through the frequency range from approximately 0 to 100 cycles per second at a prescribed rate, and at specified amplitudes. Although not of primary structural significance, shake tests are also

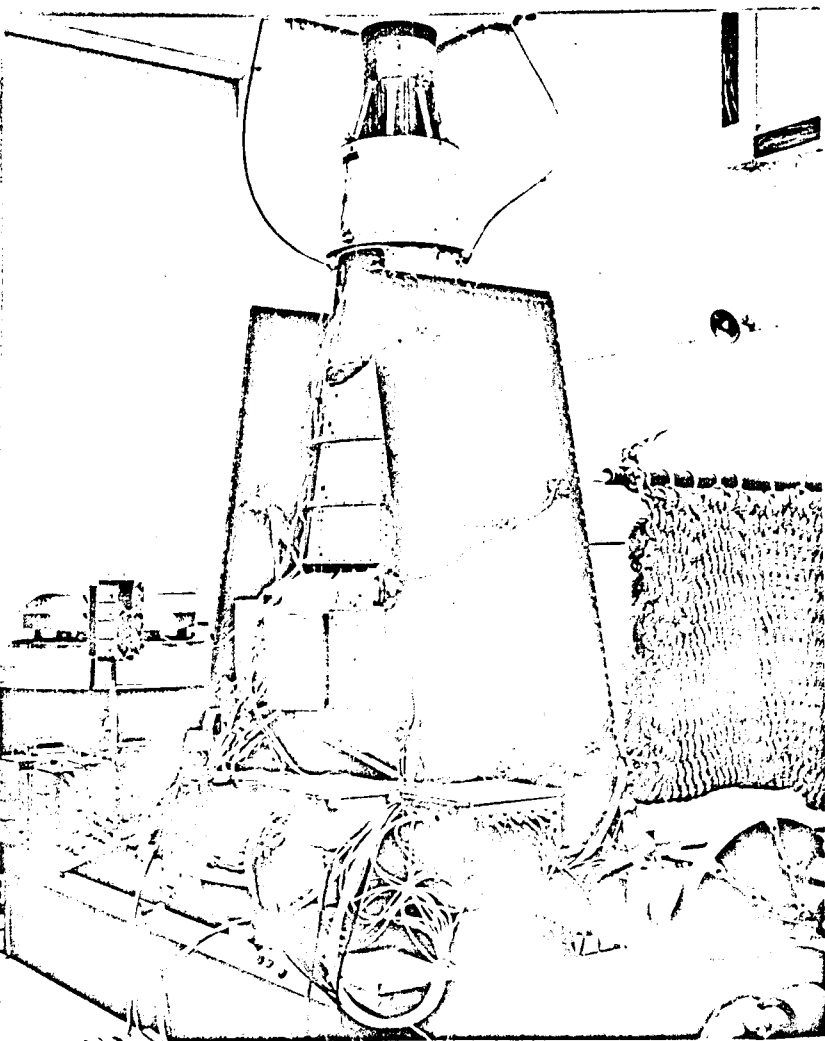


Fig. 31. Setup for forced-vibration test

conducted at higher frequencies, from 100 cycles per second to approximately 1,000 cycles per second. For such tests an electromagnetic shaker is substituted for the hydraulic shaker. The shaker system is controlled by magnetic tape.

One of the test techniques we are now using rather extensively for several purposes is that of modal vibration survey. In this technique we use small shakers singly, or in combination, to excite various primary normal modes of vibration of a test structure. Since this is a resonance technique, very little actual power is required for excitation. The spacecraft structure is mounted very rigidly on a massive structure so that characteristics of the mounting system will not significantly affect test results. One or more electromagnetic shakers are then attached at strategically selected locations on the structure, depending upon the particular mode to be investigated. A typical setup for such a test is illustrated in Fig. 32. The electronic power supply and control system for modal vibration-testing permits very precise control over the several shakers in terms of frequency, amplitude, and phase relationship. Thus, for example, two shakers may be used 180 degrees out of phase to excite a torsional mode or a higher bending mode.

Small accelerometers are attached to the structure to determine the spacecraft response to the vibration input. During early stages of a survey, one or more accelerometers are moved about from place to place to define the shape of a particular vibration mode. Once a mode has been fairly well established, several accelerometers may be attached at different locations and their outputs simultaneously recorded. In this type of test we are interested only in the overall normal modes of vibration of the structure. Such normal modes can usually be recognized by the relationship between the input excitation and the structural response. An important piece of instrumentation, therefore, is an oscilloscope, on which we display the shaker force against the shaker coil velocity. Collapse of the otherwise elliptical



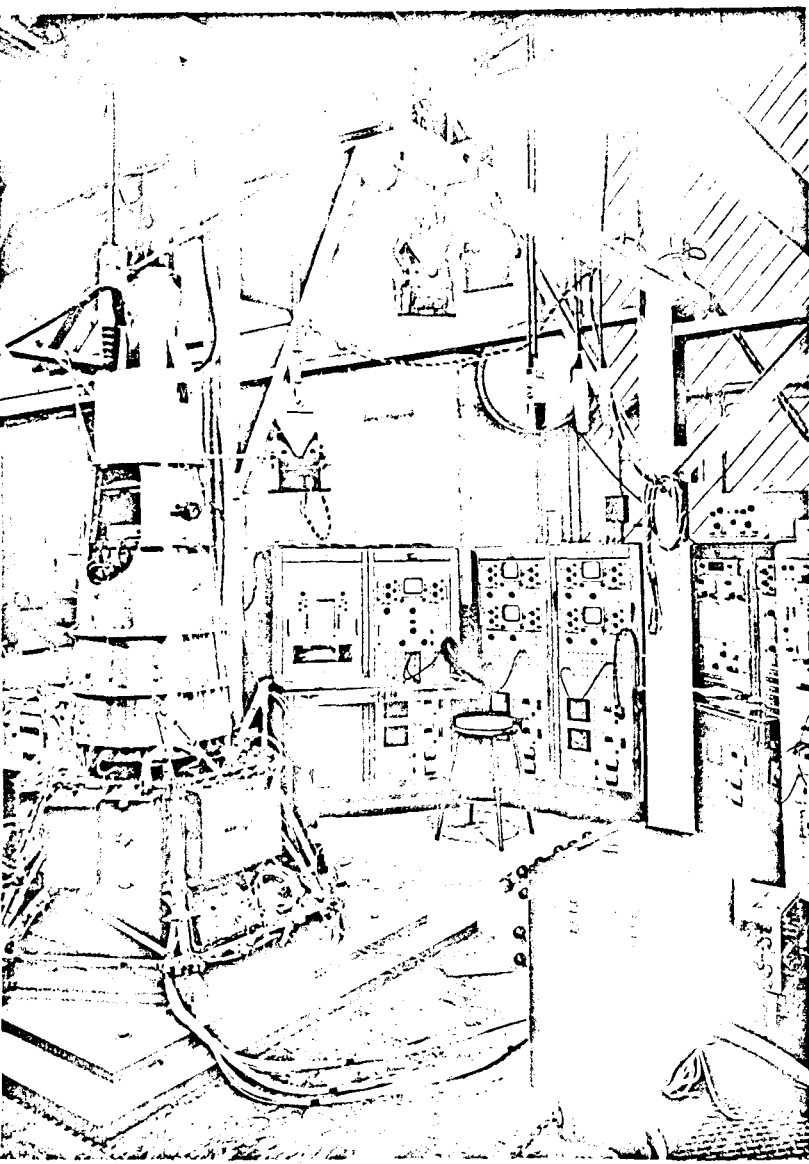
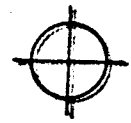


Fig. 32. Setup for modal vibration test

pattern into a line usually indicates that a normal mode is being excited.

We find the modal vibration survey to be extremely useful in several ways:

1. Normal mode shapes and frequencies can be defined and identified as a check against theoretical analysis of the structure.
2. The modal test results may be used directly as inputs to the overall structural analysis program previously discussed, since generalized-mass, stiffness, and damping matrices are derivable from test data.
3. The method is used to quantitatively determine the extent of damping present in the structure. Damping characteristics are determined through measurement of decay following

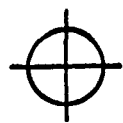
sudden termination of excitation. This application serves primarily as a verification of assumptions made in structural analysis.

4. The method has been found to be very powerful and is now used extensively as a diagnostic tool.

Modal vibration-testing is, of course, nondamaging testing, since large amplitudes and loads are completely unnecessary. At the same time, however, the modal characteristics are extremely sensitive to minor changes in linearity of the system. For these reasons, the modal survey has become a standard means for checking condition of a structure. Typically, at the Jet Propulsion Laboratory, we run modal surveys before and after all structural-qualification tests. If the resonance characteristics of the structure have not changed, this is convincing evidence that the structure has not suffered subtle damage. In some instances involving built-up structures, we have been able to detect the existence of improperly riveted joints through examination of resonance characteristics.

It might be of interest to note that the modal survey technique is currently finding application in structures of an entirely different scale than spacecraft. Dr. Donald E. Hudson and Dr. George W. Housner of the California Institute of Technology have recently conducted a modal survey of the complete steel structure of a nine-story office building currently nearing completion at the Jet Propulsion Laboratory. With very modest equipment they were able to excite five bending modes of the structure, and to obtain excellent damping-characteristic information.

As spacecraft become larger it is evident that overall "brute force" shake tests of entire vehicles will be impossible. It appears, therefore, that reliance must be placed upon a combination of analysis, resonance-type testing, and, possibly, dynamic model-testing. Although some effort has been devoted to an examination of the possibilities for model-testing, we have as yet established no basis for a confident prediction



of the significance such tests may hold in the future. It is certainly clear that the pressures for development of extremely light and extremely efficient structures will not decline in the future. Fortunately, our analytical techniques are becoming more effective and our understanding of some of the environmental factors is improving. Thus, although we can expect to face continuing design challenges, we should be able to find solutions in which confidence can be placed.

SPACECRAFT TEMPERATURE CONTROL

Within the field of engineering mechanics, perhaps the specific problem area with the smallest base of experience upon which to proceed is that of temperature control. This is because the environment encountered and the requirements imposed have not been encountered to any significant extent in engineering developments in the past. Because the environment is characterized by the absence of a gaseous medium, there is no convective heat transfer. Temperature distribution within the spacecraft itself is determined on the basis of radiation and solid-conduction heat transfer, whereas the overall or average thermal condition of the spacecraft depends exclusively on radiation.

Most Earth-based engineering systems are strongly affected by convective heat transfer, and those instances in which radiation heat transfer is of major importance usually involve radiating elements at relatively high temperatures, such as boiler tubes or reactor elements. In the case of a space vehicle, however, since convective heat transfer is absent, a much higher order of accuracy in prediction of both conduction and radiation heat transfer must be achieved if operating temperatures are to be held within reasonable limits. It has literally been necessary to develop a whole new technology for the purpose.

The basic objectives of the temperature control system for a space vehicle are very simple. The objectives are to maintain the temperatures of all elements of the spacecraft within

allowable ranges, and to do so with a minimum of added weight and consumption of electrical power. The last two requirements, at least for smaller spacecraft, generally tend to favor the use of passive techniques rather than active systems utilizing fluid transfer or mechanisms.

Environment and Conditions

As might be expected, the design problem is a complicated one, and it is very difficult to specifically allow in the design for all of the different conditions which prevail during all phases of the space flight. Accordingly, the normal approach is to give primary attention to the situations under which steady-state temperatures will be achieved and then to examine the more transient conditions encountered during maneuvers to identify those which have a critical effect upon the equipment temperatures. If such a critical condition is found, a specific solution is then introduced.

Cruise Phase:

Let us consider the conditions relating to the temperature control problem which prevail during the long cruising flight of the spacecraft. Generally, the time rate of change of any parameter is small enough that the spacecraft is essentially in equilibrium at all times. The average thermal condition of the spacecraft will be determined by a balance between the thermal energy absorbed from the Sun and the thermal energy radiated from the spacecraft to outer space. The amount of nonthermal energy radiated is negligible.

The spacecraft described so far in this paper are of the attitude-stabilized variety, specifically in relationship to the Sun. Thus, the only significant source of external energy is always in the same position relative to the spacecraft. This is, of course, an important consideration in design. For interplanetary missions the intensity of the energy received from the Sun varies substantially throughout the flight, as illustrated in Fig. 33. For a typical mission to Venus the radiation intensity increases by approximately a factor of 2 over a flight time of

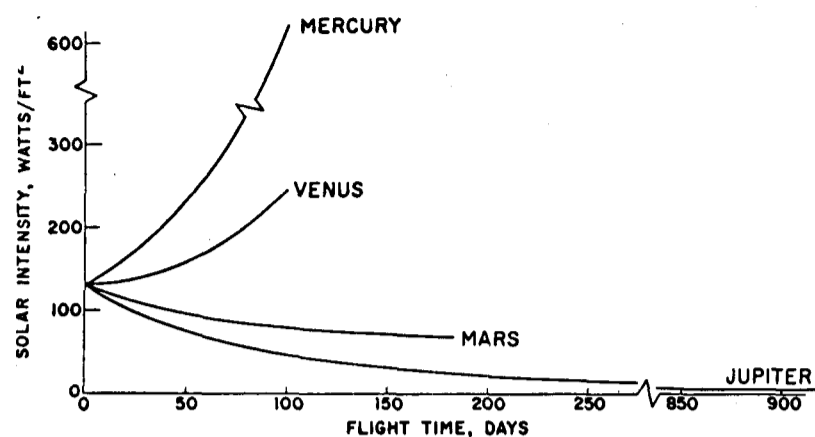


Fig. 33. Variation of solar radiation intensity with time for several interplanetary missions

approximately 110 days. For a Mars mission, the radiation intensity decreases by a factor of approximately $2\frac{1}{2}$ in about 200 days. It is to be noted that the corresponding variation in radiation intensities for Mercury and Jupiter are even more severe. Such large variations in the energy received by the spacecraft nearly preclude the use of totally passive systems for equipment with normal operating limits.

Temperature control design is also affected by configuration changes required during the cruise phase. A typical example of a configuration change is that of the high-gain antenna, which must for most missions change its attitude relative to the spacecraft in order to keep the Earth in view. For example, the Mariner 2 flight required antenna angles ranging over approximately 120 degrees. This introduced an interesting problem in the temperature control design for the sensitive sensor system used to maintain Earth lock. Another element of variation during cruise flight corresponds to changes in electrical power for individual items of equipment as they are turned on and off in connection with science measurements or impending maneuvers. As will be noted later, in some instances it is possible to arrange the design so that such variations have little effect on the temperature control problem whereas in other cases involving isolated items, it is necessary to introduce heating circuits to dissipate the equivalent equipment power during periods of nonoperation.

The midcourse maneuver, which is typically required during cruise flight, introduces

two conditions that affect temperature control: (1) a change of orientation relative to the Sun, so that the thrust vector will be oriented in the proper direction, and (2) heat dissipation from the rocket motor used to provide the necessary momentum change. Both of these problems must be treated, but in any case maximum advantage is taken of the heat capacity of the spacecraft, since the time durations are short.

Other Phases:

Let us now consider some of the conditions during other portions of the flight that affect spacecraft temperature control. First of all, we have the ground-conditioning problem. On the ground the spacecraft is enclosed within the launch vehicle nose cone for several days prior to flight. During this time extremes of thermal input to the shroud can be expected, so some scheme must be provided for air-conditioning. One method is to introduce conditioned air from a ground system. However, it is sometimes difficult to provide adequate ducting into the shroud enclosure. For this reason, a cooling blanket was used on the Agena vehicle. The cooling blanket surrounds the shroud and is provided with conditioned air from a ground source. The blanket is removed by a lanyard system at the moment of liftoff.

As the launching rocket rises through the Earth's atmosphere, high velocities are attained at which very high stagnation temperatures occur over the forward portion of the shroud. To prevent excessive heat-transfer rates from the hot shroud to the spacecraft during this portion of flight, the Mariner 2 shroud was provided with an inner radiation shield, as illustrated in Fig. 34. The liner also served as a bumper guard for the spacecraft to avoid hang-up during shroud ejection. The liner was supported by insulating posts from the main shroud. The large difference between shroud and liner temperatures is to be noted.

The shroud is usually ejected approximately at the altitude at which the parking orbit will be established and prior to acceleration of the vehicle to orbital velocity. Thus, the spacecraft is not protected against atmospheric heating while

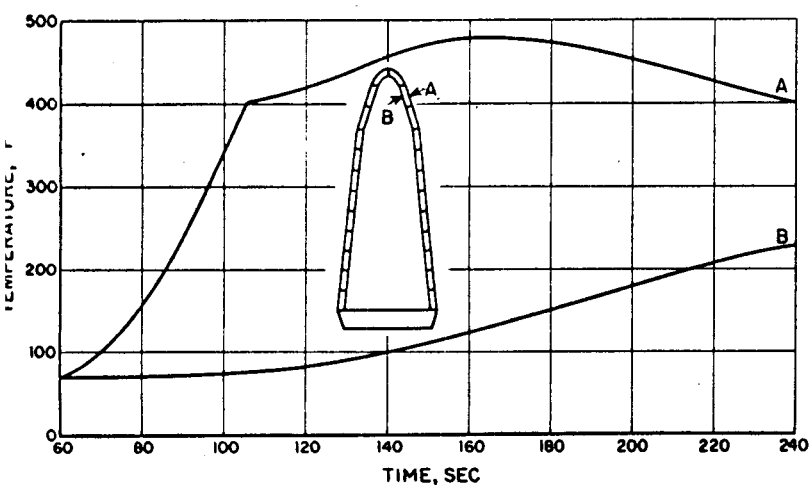


Fig. 34. Effect of shroud liner on heat transfer

in parking orbit. Frequently it is true that for a given launching rocket, the lower the parking orbit, the greater the payload weight which can be injected. However, the lower the parking orbit, the greater is the atmospheric heating on the spacecraft. Figure 35 illustrates the variation in heating rate as a function of parking-orbit altitude. The minimum permissible parking orbit will be determined by the tolerable spacecraft heating.

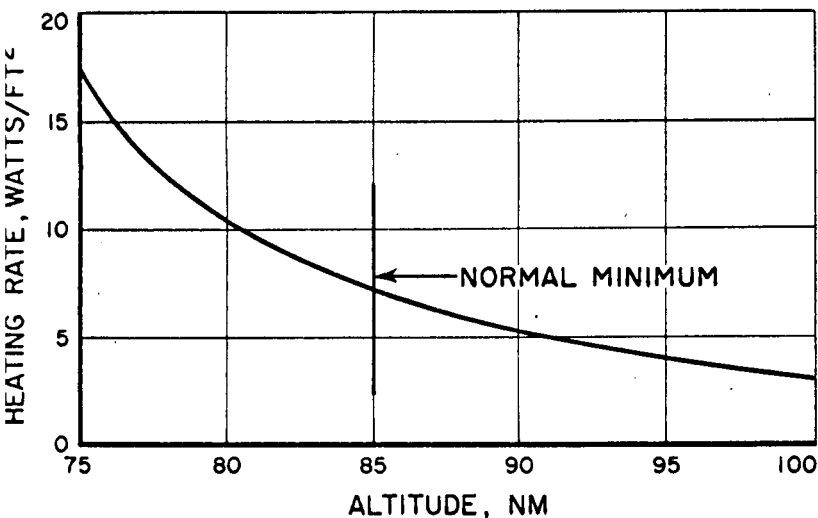


Fig. 35. Atmospheric heating rate as a function of altitude at satellite velocity

At the other end of the flight mission, we encounter special conditions associated with approach maneuvers or other constraints imposed by mission requirements. Even though the terminal phase may not require a change of spacecraft attitude (as in the case of the Mariner 2 mission), there will usually be significant changes in the mode of internal operation during this phase.

For example, all of the encounter instrumentation, which has been going along for the ride during the cruise portion of flight, will now be turned on to serve the purposes of the mission. For many cases, a change of orientation will be required, thus establishing an entirely new Sun input, and for certain types of close flyby missions the energy reflected from the target planet may also be significant. Figure 36 illustrates the change in thermal power which occurs in the Ranger vehicle, currently under development at the Jet Propulsion Laboratory, as the spacecraft approaches the Moon and establishes an orientation appropriate for television picture-taking. It is seen that a large change in the thermal situation occurs. In the Ranger, the situation is handled through a combination of design accommodation and thermal heat capacity.

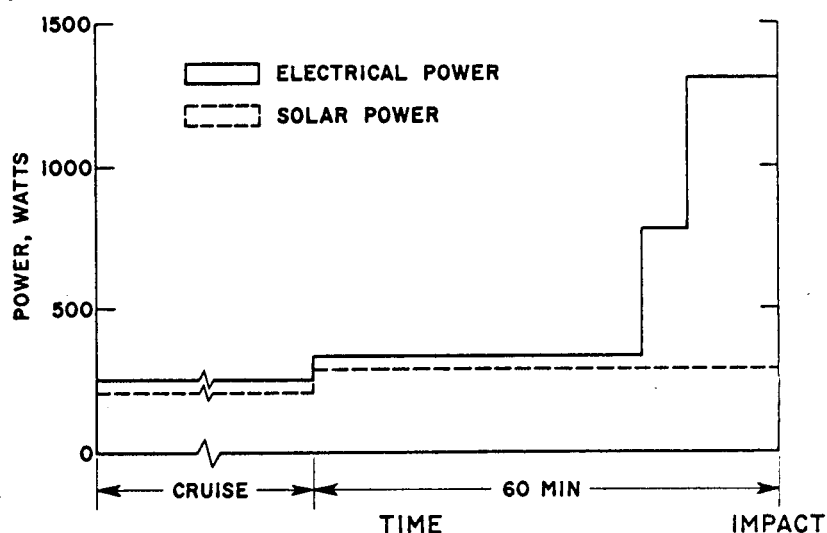


Fig. 36. Typical thermal power changes near target encounter

Three special mission cases deserve mention, but will not be discussed in any detail. A planet-orbiter will be strongly affected by either its changing attitude with respect to the Sun, or with respect to the planet about which it orbits. In either case, special temperature control problems will be encountered. In a planetary-atmosphere entry vehicle, a major problem of atmospheric heating is encountered. Basically, the extremely high kinetic energy which the spacecraft possesses by virtue of its velocity must be dissipated as the vehicle enters the planetary atmosphere. Figure 37 illustrates the magnitude of this problem, and

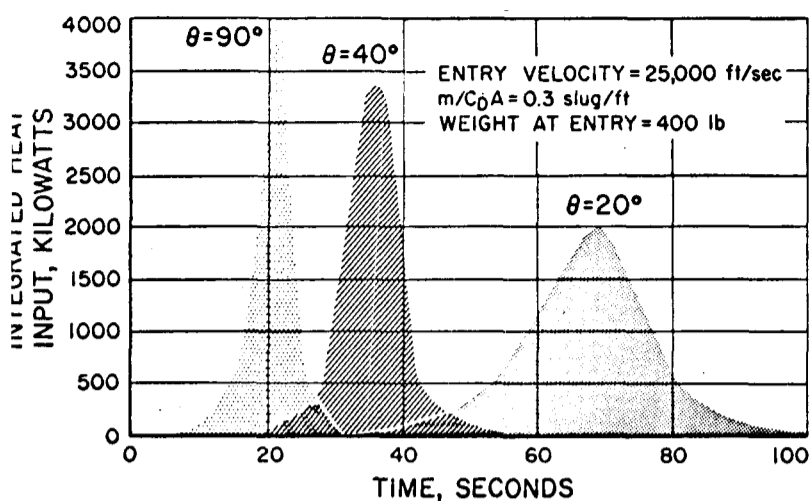


Fig. 37. Atmospheric entry heating for typical Mars case

indicates the variations which can be expected, depending upon the angle of entry. The case illustrated corresponds to a "capsule" of 400 pounds, suitable for entry into the atmosphere of Mars. Obviously, very special means must be taken to cope with the high thermal powers involved. A space vehicle intended to land on the surface of another body must operate in a very complex thermal environment, which may or may not involve an atmosphere providing convective heat transfer. For such a case, in addition to direct solar heat input, we must deal with both spectral reflections and infrared radiation associated with surface features. It is highly unlikely that a purely passive design will suffice in such a situation.

Design Approach

As mentioned earlier, the basic approach to vehicle design for temperature control is generally developed from a consideration of conditions during the cruise phase. Transient conditions are then considered to determine whether or not they have a critical effect on any spacecraft equipment. For a planetary flight, because of solar-intensity variation, average spacecraft temperature can be expected to increase or decrease depending upon whether the mission is toward or away from the Sun. Thus, unless active temperature control systems are utilized, we can generally expect spacecraft temperatures to increase or decrease during the mission, as illustrated in Fig. 38. If the predicted

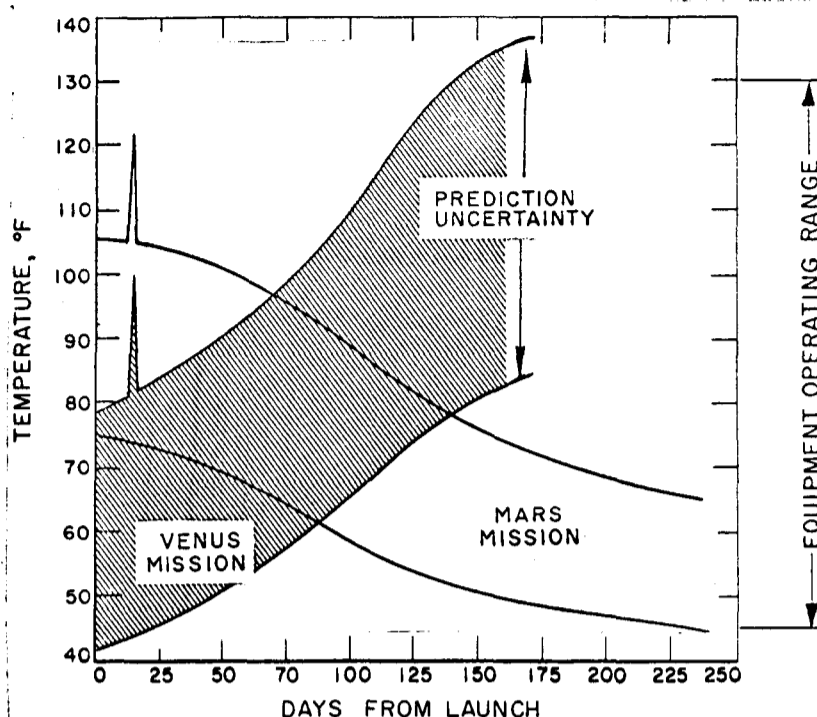


Fig. 38. Anticipated temperature variations and uncertainties for planetary spacecraft

temperature variations are in excess of equipment limits, active devices may be introduced, but only to the extent required to reduce the temperature variation to an acceptable range. This Figure also illustrates the substantial uncertainty in temperature prediction which results from a number of sources.

The equilibrium thermal condition of the spacecraft at any time will be determined by a balance between the total energy absorbed from the Sun, and the total energy radiated to space. Average temperature, then, can be controlled within practical limits by proper selection of coating materials applied to surfaces exposed to the sunlight and to surfaces radiating to space. The characteristics of some surfaces are actually determined by other conditions. For example, a large portion of the area exposed to the Sun is covered by solar cells. These cells are selected primarily for their energy-conversion characteristics, and only limited control over absorptivity is permissible using filters.

The spacecraft system includes a large number of different subsystems and assemblies. Many items of equipment operate intermittently, or have variable power-loading curves. Accordingly, temperature control is accomplished most readily by grouping as many items as possible

into one thermal region, thus permitting an averaging of the thermal loads to take place. Of course, this has the concurrent disadvantage that the allowable temperature range for a group of devices is generally smaller than for any one device, since the maximum and minimum temperatures allowed for the group correspond to the most extreme item at each end of the range. Those items of equipment which cannot be included within the one controlled region are generally treated individually in an isothermal manner; that is, no attempt is made to rely specifically on conduction of heat through connecting structure.

This general approach and the important thermal considerations are illustrated in Fig. 39, which shows the thermal-balance situation prevailing for the Mariner 2 spacecraft with respect to the main equipment enclosure. It should be recalled at this point that the Mariner 2 carried out a flight to Venus, and that the intensity of solar radiation increased by a factor of 2 during the flight. To reduce the increase in spacecraft temperature during the flight, an attempt was made to make the design as insensitive as possible to direct solar input. The top of the equipment enclosure was insulated by a blanket consisting of many layers of very thin mylar, each coated with vapor-deposited aluminum. Over this blanket was a layer of teflon having a mirrorlike vapor-deposited layer of aluminum on its under surface. This combination resulted in the following characteristics:

1. Minimum absorption of solar energy (reflection from the mirrorlike teflon coating)
2. Relatively high infrared emission of energy from the outer surface of the teflon
3. Minimum heat conduction through the blanket into the spacecraft

The lower surface of the enclosure was covered by a radiation shield with a low-emissivity surface in order to minimize the heat losses to space from this part of the enclosure, and confine primary heat rejection to areas which could be readily analyzed and controlled.

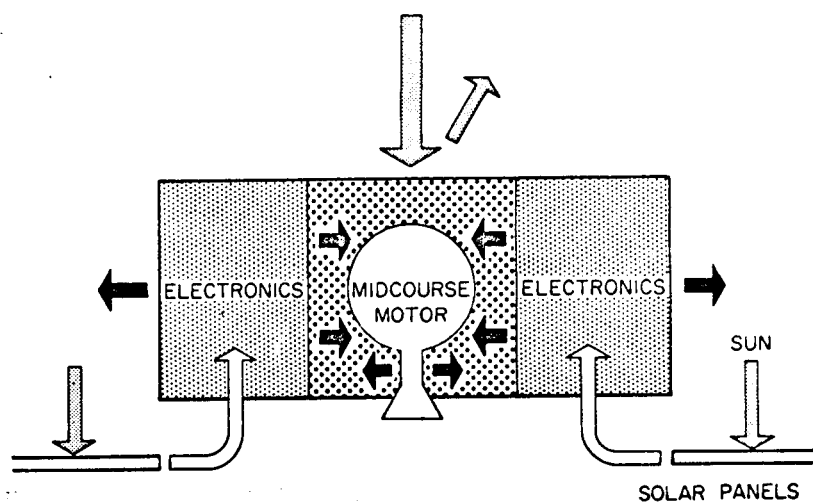


Fig. 39. Schematic of Mariner 2 equipment enclosure temperature control

The major source of energy input to the enclosure during cruising flight was provided by electrical energy from the solar panels. This electrical energy was ultimately dissipated either in items of equipment or in voltage-regulation devices, both types of which were located within the enclosure. Thus, the thermal energy was nearly constant, although its detailed distribution was not. Primary design control over the enclosure temperature was provided on the sides, where white-paint patterns were superimposed on low-emissivity surfaces to give the proper heat rejection. All internal surfaces within the enclosure were painted black to achieve as much uniformity as possible. The rocket motor, which operated only for a brief period during the mid-course maneuver, is shown in Fig. 39. Steps were taken to shield the equipment bay from the rocket nozzle in order to lengthen the time during which the energy generated was transmitted to other parts of the spacecraft.

With the preceding description of the heat-balance situation for a spacecraft equipment enclosure in mind, Fig. 40 indicates the manner in which the thermal control design may be approached. The spacecraft illustrated incorporated a hexagonal structural enclosure for most equipment items. For purposes of design analysis, this enclosure was thought of in terms of six compartments. Within practicable limits established by other design requirements, high- and low-energy dissipating items were distributed

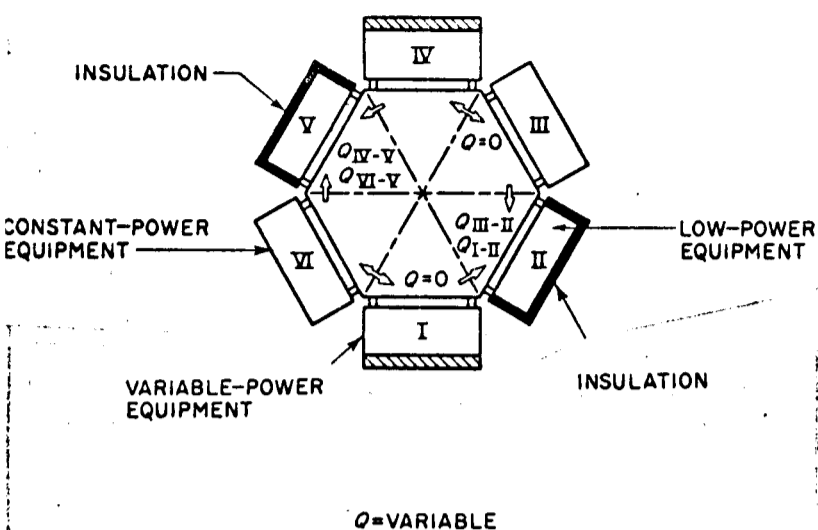


Fig. 40. Temperature control approach for hypothetical spacecraft

throughout the enclosure so as to minimize net heat transfer from one bay to another. The following three situations are illustrated in Fig. 40:

1. Two bays included only relatively low-power consumption equipment. The external surfaces for these bays were insulated, and the outer layer of insulation was coated with a material of low emissivity. Despite this treatment, however, some net heat flow from the adjoining bays was required to maintain the proper temperature.
2. Two bays incorporated equipment of average power consumption and of nearly constant energy dissipation throughout the flight. The external surfaces for these bays were coated in a combination of high- and low-emissivity materials so as to balance total heat rejection against power dissipation. This pattern was not necessarily the same for both bays.
3. Two bays incorporated equipment with variable power dissipation during the flight. Because of this variable power dissipation, it was necessary to provide a compensating variable-heat-rejection mechanism. The mechanism illustrated is of the "venetian blind" variety.

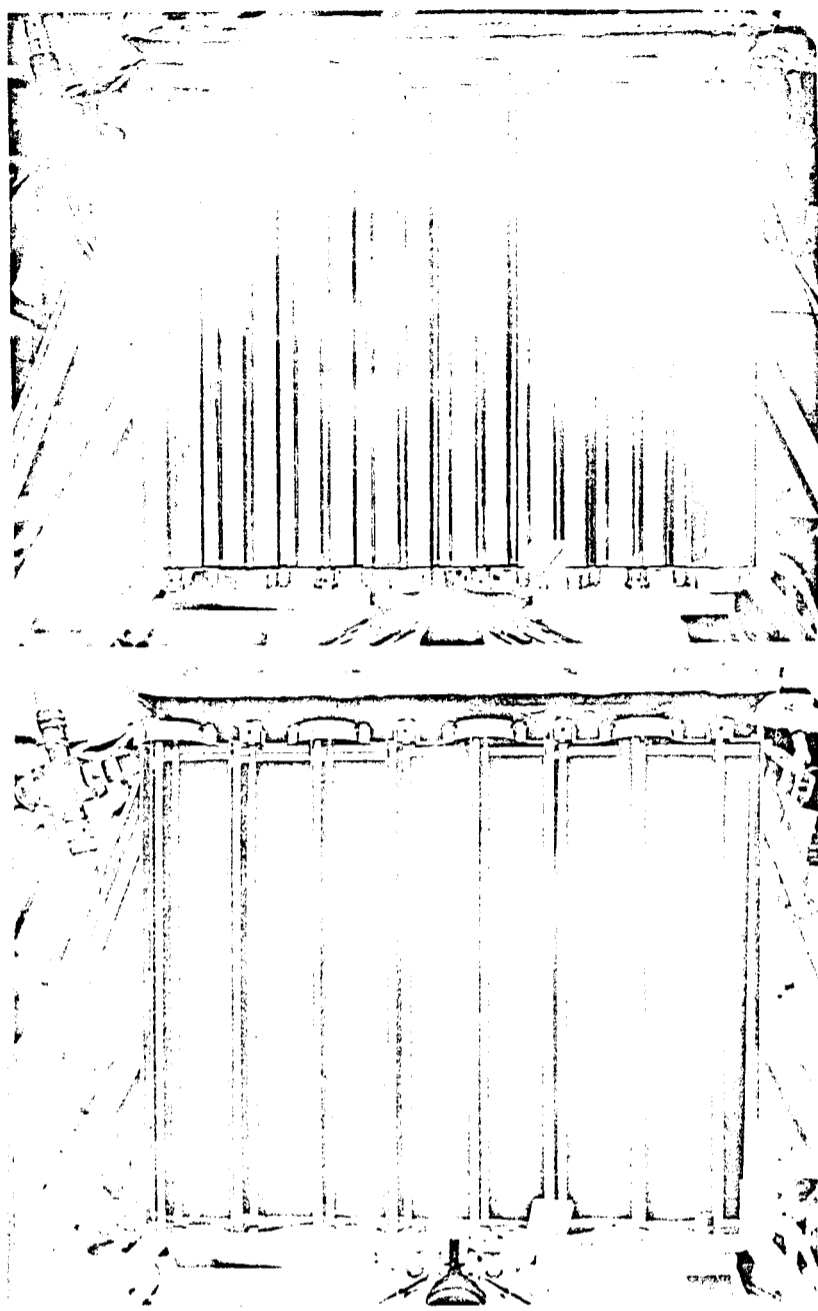


Fig. 41. Louver-type radiation control device

Several devices have been designed and developed for automatic control of radiation emission. Figure 41 is a photograph of the venetian-blind or louver-type device. It consists of a series of louvers or slats, each of which rotates on bearings about a lengthwise axis and is actuated by a bimetallic element. At the low end of a calibrated temperature range the slats are closed, as illustrated in the top half of Fig. 41. The outer surface is of polished aluminum, having a low emissivity. At the high end of the temperature range the louvers are open, as illustrated in the lower half of Fig. 41, exposing the structural surface which is coated with a high-emissivity material. The entire device for covering an area

of approximately 180 square inches weighs approximately 2 pounds. At 100°F the difference in heat rejection between full-open and full-closed positions is 40 watts.

Another device for controlling temperatures in spot locations is illustrated in Fig. 42. This device is rather similar in concept, but different in mechanization.

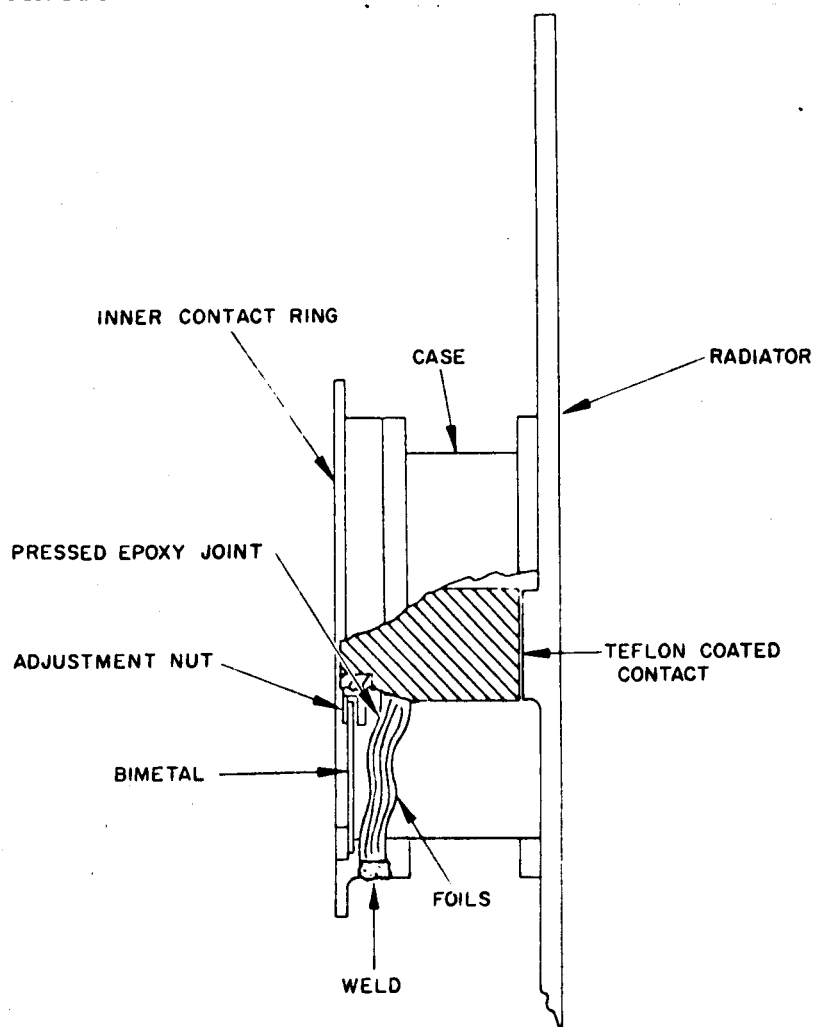


Fig. 42. Button-type radiation control device

Analytical Tools

Although analysis is required in establishing the thermal control design for spacecraft, its application is limited for several reasons. Reasonably valid treatment can be given to conduction heat transfer through solid materials. However, when mechanical joints are involved, there is considerable uncertainty as to the relationship between heat transfer and definable physical parameters for the joint. This is particularly true under the hard-vacuum conditions of space. Conduction analysis, therefore, is primarily valid in situations where mechanical joints are not an important factor in the heat path. In many

situations the predominant heat-transfer mechanism is radiation, primarily infrared, and at relatively low temperatures. Unfortunately, detailed analysis of a complex structure becomes almost impossible because of the mathematical complexity involved in applying basic principles to complicated shapes. As is well-known, the determination of "view factor" between two bodies is a complicated calculation even for very simple shapes, such as two simple cylinders. So far, it has appeared to be impracticable to perform detailed calculations on the highly complicated multiple-element structures with which we typically deal in a spacecraft. Another difficulty has been the dearth of good information concerning the radiation properties of coating and surface materials. Even in those instances where a mathematical model can be formulated and treated numerically, substantial uncertainties in temperature prediction have remained because of the coating problem.

As a consequence, theoretical analysis has been used primarily in three ways:

1. To provide, through approximate analysis, a reasonable basis upon which to establish a basic design concept,
2. To determine paint patterns on external surfaces of equipment enclosures having specified energy-dissipation requirements and temperature limits,
3. To interpret experimental data and determine adjustments to design conditions for which experiments have been made.

Thus, it is seen that analysis is an important tool but cannot be relied upon by itself to establish design confidence. The final design decisions arise out of a blending of the analytical, experimental, and previous flight results as they become available. All three are required for confidence. It is worth noting, however, that techniques of analysis and implementation through computer programs are becoming more powerful and more useful.

Experimental Methods

At the present time, fairly heavy reliance must be placed upon experimental means for establishing and verifying the temperature control design of a spacecraft. Facilities of all types are employed, ranging from very simple laboratory setups to very large and elaborate space simulators, incorporating simulated solar energy sources. Tests conducted range from detailed investigation of surface-coating properties to thermal-balance and temperature-distribution tests for entire spacecraft.

As mentioned earlier, one of the sources of uncertainty in prediction of spacecraft temperatures derives from the lack of information on coating characteristics. We have found it necessary at the Laboratory to conduct fairly extensive investigations of coating materials to establish values for use in analysis and design. Figures 43 and 44 illustrate experimental samples and the test setup used in determining absorptivity for a number of such materials. In this particular test device seven small disc samples of coating materials are mounted on the surface of a fixture. Heater elements are attached to the rear surface of each sample and very careful attention is given to insulation of the samples from the mounting fixture and from each other. The fixture is installed in a vacuum chamber with a quartz window on the front, as shown in Fig. 44, and energy is supplied to the heater elements to bring each element to a desired nominal temperature. When all temperatures are stabilized, a simulated solar-light source is turned on, and the change in heater power required to hold the same temperature on each sample is determined. Thus, the change in heater power is a direct measure of the radiant energy absorbed through the surface.

Development of spacecraft has required not only the parallel development of launch rockets, launch facilities, and tracking networks, but has also required the development of space simulators. A space simulator essentially consists of three elements: a vacuum chamber capable of providing and maintaining a vacuum level of 10^{-5} millimeters of mercury or better, a cold-wall liner

cooled to approximately the temperature of liquid nitrogen or colder, and a highly collimated beam of light to simulate energy from the Sun. The first two features are relatively easy to provide; the last feature, solar simulation, is usually very difficult and expensive. For this reason, a considerable amount of the development work on spacecraft so far has been accomplished with space simulators providing only the vacuum and cold-wall conditions.

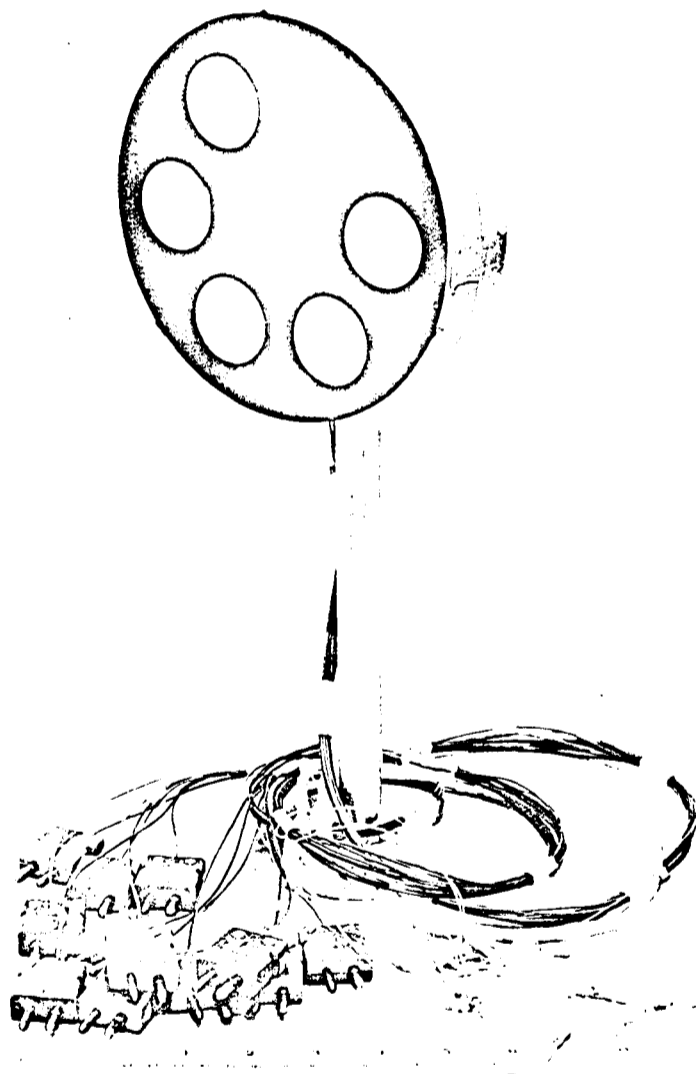


Fig. 43. Coating samples in fixture for absorptivity tests

The cold-wall vacuum technique is based upon the assumption that the amount of energy absorbed by surfaces exposed to the Sun can be adequately estimated. The corresponding energy is then supplied through electrical heaters applied to corresponding portions of the spacecraft structure. Figure 45 illustrates a spacecraft test



1. Integrated intensity equal to that

might be expected, the development of the simulator posed almost as many engineering problems

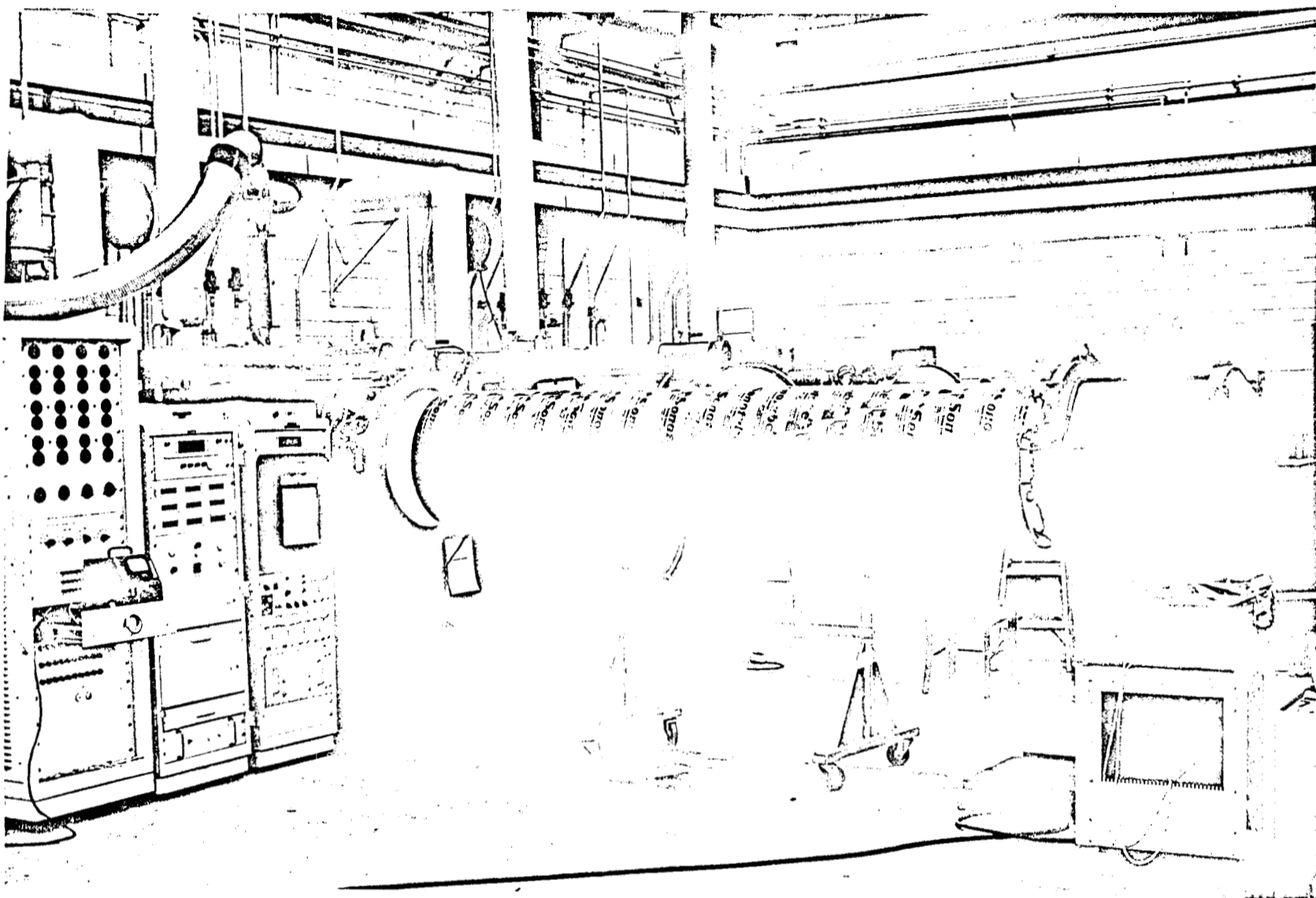


Fig. 44. Setup for absorptivity tests

prototype with heaters applied for this purpose. The metal plate to which the heater elements are attached is cut out to eliminate shadowed areas and serves to distribute heat energy over sunlit portions of the spacecraft. In test, insulation was applied over this plate to reduce heat losses to the chamber walls. Note that with this technique, the radiation and conduction heat transfer within the spacecraft in a vacuum environment is simulated, and heat loss to space as represented by the chamber cold walls is duplicated. The method has been used with some success, but is limited because of the simplifying assumptions that must be made.

For true simulation of space conditions, solar simulation is required. The solar simulator should have the following characteristics:

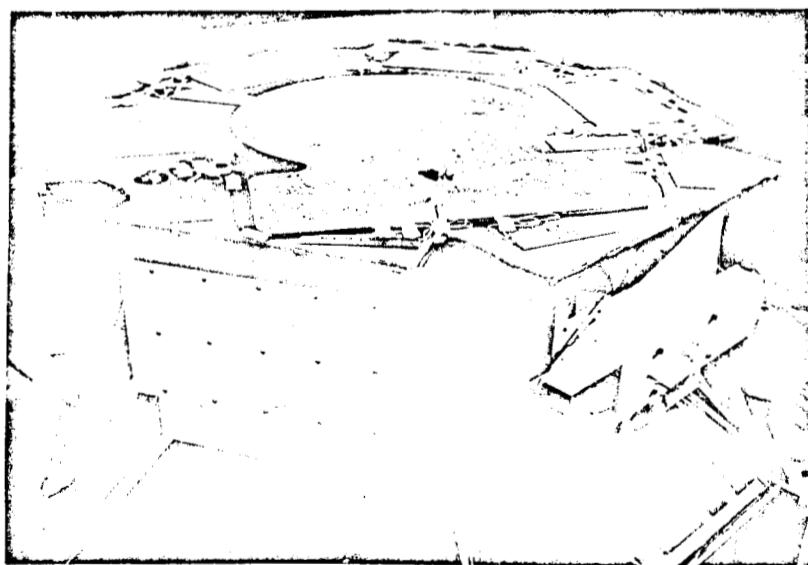


Fig. 45. Mariner 2 prototype with heaters for cold-wall vacuum temperature tests



1. Integrated intensity equal to that which will be experienced in space flight. For Venus missions this implies a source capable of providing approximately 260 watts per square foot.
2. An illuminated area the size of the spacecraft to be tested.
3. Well-collimated light, comparable to sunlight. This is primarily to achieve shadow conditions comparable to those to be experienced in flight.
4. Uniformity of energy distribution throughout the illuminated test volume within approximately ± 5 percent of nominal.
5. Reasonable duplication of the solar energy spectrum.

Although it may not be obvious from the foregoing list, it can be stated that it becomes extremely difficult and expensive to meet the above conditions. The development of simulator technology has been paralleling that of spacecraft, and simulators are by no means leading the spacecraft. As might be expected, it has been necessary to accept compromises in simulator characteristics in order to stay within the simulator state of the art and acceptable cost limits.

At the Jet Propulsion Laboratory we now have a selection of space simulators of differing sizes and capabilities. The largest of these simulators is 25 feet in diameter and approximately 30 feet high. It is capable of providing a vacuum of 10^{-6} millimeters of mercury and is equipped with a full cold wall cooled with liquid nitrogen to approximately 100°K. At the present time we have installed in this chamber a solar-simulation system which provides well-collimated light of good uniformity in approximately a 5-foot beam. Maximum intensity is approximately 170 watts per square foot--somewhat greater than solar intensity at the orbit of Earth. An improved light system is currently under development which will provide a larger beam of higher intensity.

The 25-foot space simulator with a JPL spacecraft installed is shown in Fig. 46. As

might be expected, the development of the simulator posed almost as many engineering problems as did the development of the spacecraft. Figure 47 shows a prototype of the Mariner 2 spacecraft installed in the solar-simulation light beam. Actual development tests are conducted in vacuum. The photograph was taken during a preliminary investigation in which incident energy on the spacecraft was being mapped. The extreme complexity of the spectral-reflection situation from miscellaneous items of structure is evident in this photograph. This illustrates one of the reasons solar simulation is an important element of a space simulator.

Although it appears that we have not yet suffered a mission failure caused by inadequacy of temperature control design, our predictions of operating temperatures have been considerably less accurate than we would like. Figure 48 shows typical temperatures experienced by equipment on the Mariner 2 spacecraft during its flight to Venus,

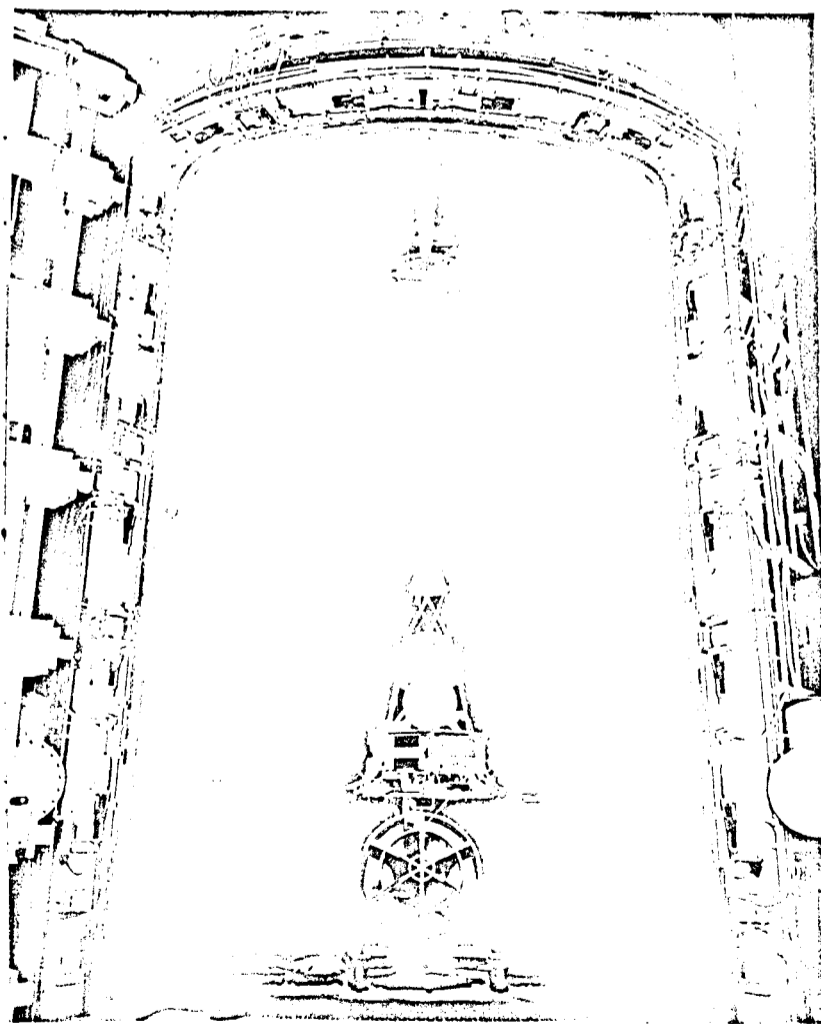


Fig. 46. 25-foot space simulator at the Jet Propulsion Laboratory



Fig. 47. Mariner 2 prototype in simulated solar light

as compared with our predictions. It is to be noted that the temperatures started out at a considerably higher level than we had expected. Consistent with this, the temperature rise during flight was also higher than predicted. One reason for this was that the control louvers, which had been expected to be closed in the early stages of flight, were already partly open, so that their compensating effect was greatly reduced.

Although we have carried out many analyses and conducted many tests, we have still not been able to fully account for the observed high temperatures. Among the factors which certainly contributed are errors in absorption prediction, spectral reflectance from structure, and deterioration of emitting surfaces during flight. However, no combination of these factors has been identified in sufficient magnitude to explain the observed discrepancy. This simply illustrates the point that

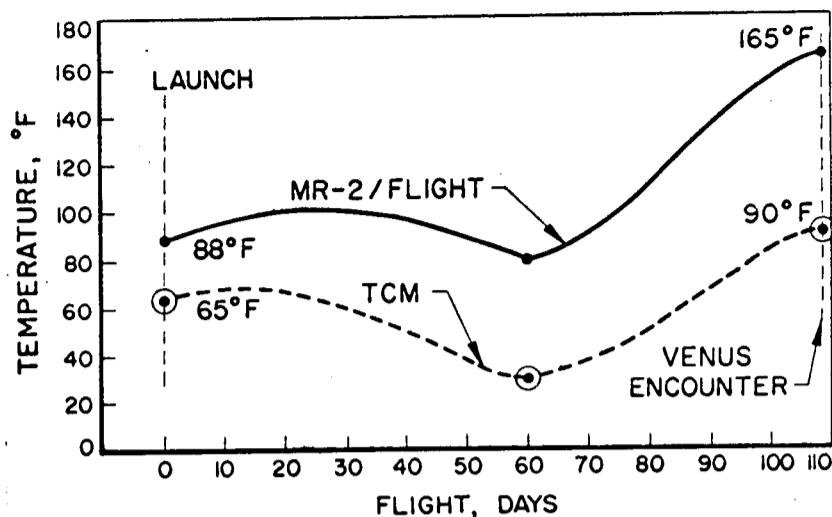


Fig. 48. Mariner 2 flight temperatures compared with predictions

there is a great deal yet to be learned about temperature control of space vehicles and a great deal of technology yet to be developed.

CONCLUSION

The technology of space flight is in its infancy. In a few areas, such as temperature control, new problems have been attacked, but we are still largely dependent upon methods and approaches developed for aircraft and missiles. This can only remain a temporary situation, for the demands of space are new and rigorous. New materials, new design concepts, and new methods of analysis and test will be required to achieve the levels of performance upon which practicable future space missions depend.

Within the scope of this discussion, it has been possible only to summarize the nature of problems encountered in a few selected areas of space vehicle development and to describe the current status of corresponding technology. It must be recognized, however, that this technology is changing rapidly--so rapidly, in fact, that it is sometimes difficult to define what is "current". Thus, if the methods and approaches I have described have in some instances appeared fuzzy, this may simply reflect the blur of rapid motion.

Looking to the future we can see a whole new array of problems. Some of these problems will arise out of new and more ambitious mission objectives involving such things as landing and

roving vehicles; others will be introduced with the development of major new systems such as nuclear power and electric propulsion. Adequate design solutions to the problems will be found, and in the process we will develop a true space technology.